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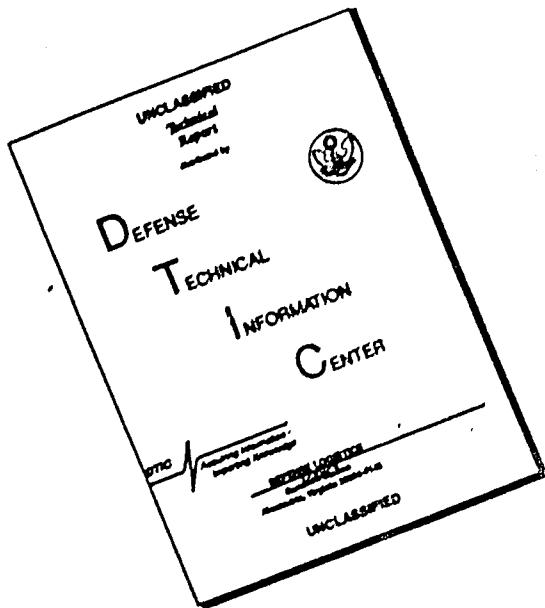
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DETAILED FINAL REPORT OF RESEARCH ON
HIGH-SPEED ROTARY-FIXED WING AIRCRAFT

VOLUME VII

SAMPLE AIRCRAFT POWER PLANT AND FLOW ANALYSIS

OFFICE OF NAVAL RESEARCH, AERONAUTICS BRANCH
PROJECT NR 250-COI CONTRACT NO. NR-250-COI

Report 1905-1
20 December 1956

Serial 15

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*Enclosure (5) to
MAC Rept. 2136-70 P-1756*

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DETAILED FINAL REPORT OF RESEARCH ON
HIGH SPEED ROTARY-FIXED WING AIRCRAFT

VOLUME VII

SAMPLE AIRCRAFT POWER PLANT AND DUCT ANALYSIS

SUBMITTED UNDER Contract NDonr-54801 to the Office of Naval Research,
Amphibious Branch, Project NR 24-1-001

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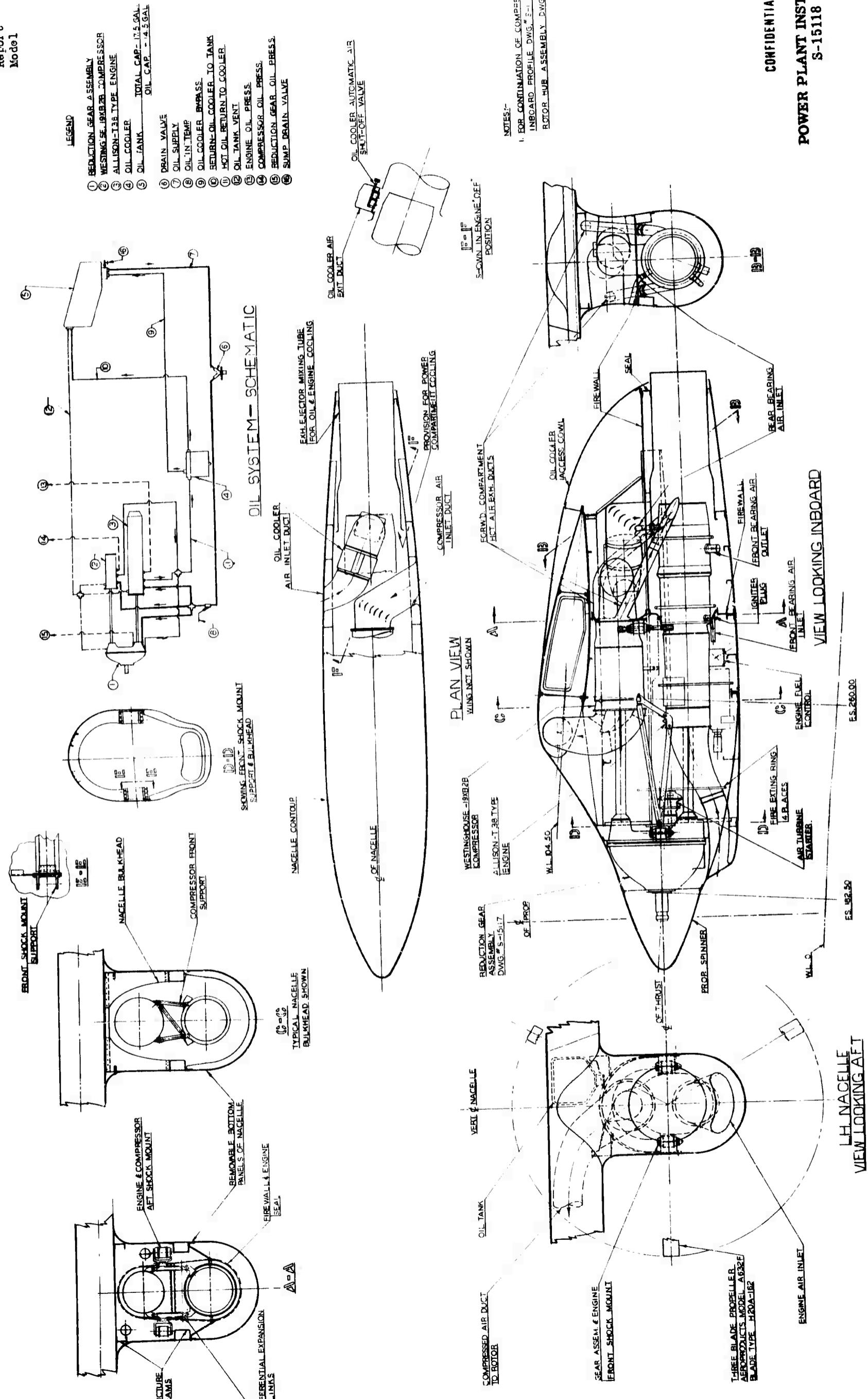
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PAGE 1REPORT 1905MODEL 781. GENERAL POWER PLANT DESCRIPTION

1.1 General - The MAC Model 78 is provided with two gas turbine engines which drive either compressors for furnishing air to the pressure jet driven rotor or tractor propellers for high speed forward flight. A power unit nacelle is mounted on either side of the fuselage.

1.2 Engine - An Allison Model 501 gas turbine power section is mounted in each nacelle supplying shaft power to the propeller and compressor. A cooling air ejector is fitted to the engine exhaust pipe, drawing cooling air over the engine compressor section, combustion section, and oil radiator.

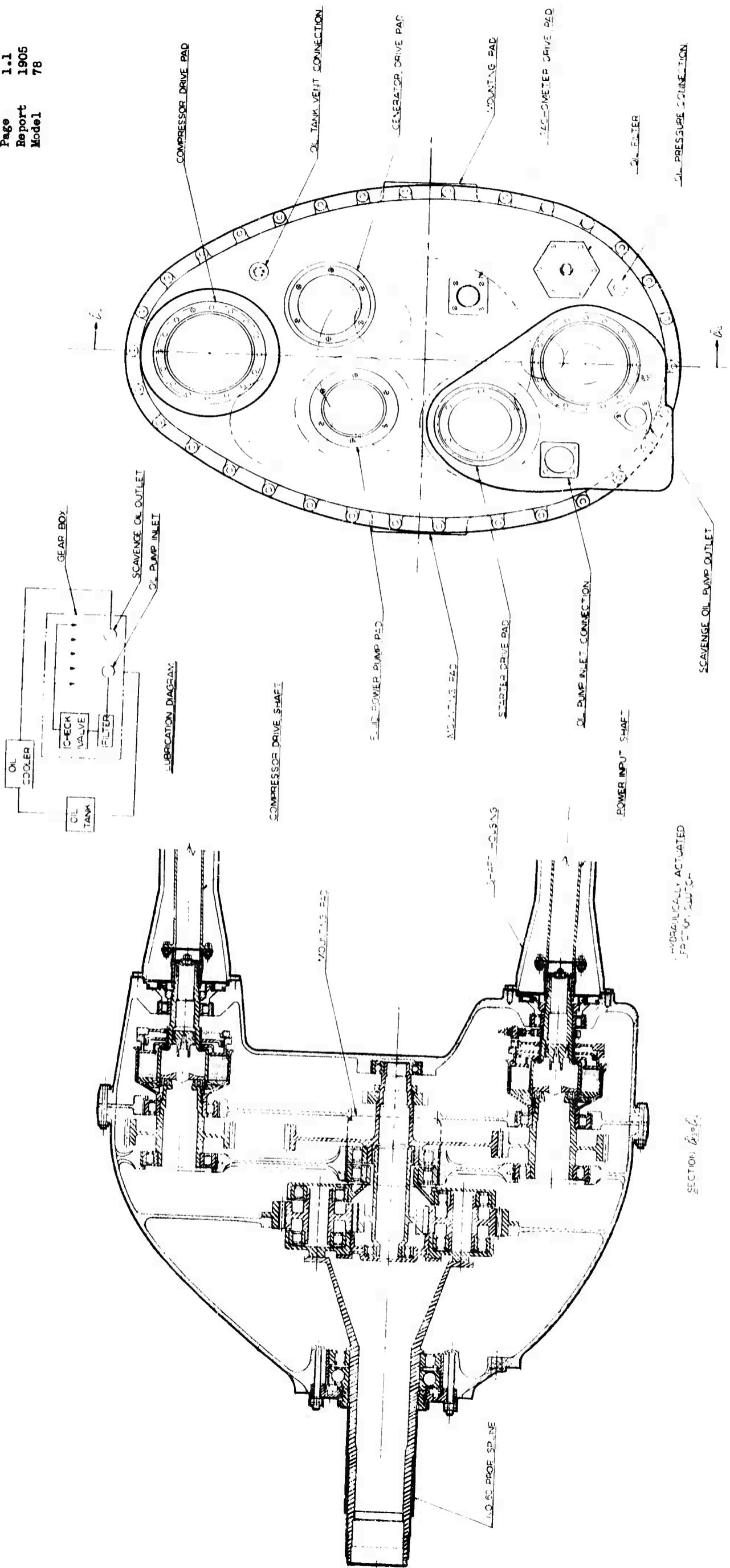
1.3 Pressure Jet Compressor - A Westinghouse 19XB axial flow compressor is employed to supply compressed air through a duct system to the rotor tip burners. This compressor is mounted directly above the engine with the compressor operated at engine speed by means of a drive shaft connected to the turbine shaft flange at the final discharge stage of the compressor.

1.4 Propeller - An Aeroproducts Model A632F propeller is mounted in the nose of each nacelle. During normal propeller operation, the propeller pitch is controlled by the engine control system. When the compressor is engaged to the engine, the propeller is held at the pitch resulting in minimum power absorption.

1.5 Gear Box - The engine drives the propeller and axial flow compressor by means of a modified Allison XT-38 gear box. The compressor shaft rotates in the opposite direction from the engine drive shaft at the same speed as the engine. Also included in the gear box section is one clutch permitting the engine to be disengaged from the gear box, and a second

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TRANSMISSION ASSEMBLY
S-15117

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3. ENGINE RPM 14,300 PROP SHAFT RPM 1600
2. OVERALL REDUCTION 1:7.95
1. FIRST STAGE PLANETARY GEAR REDUCTION RATIO SAME AS THAT
OF XTB-38 PLANETARY GEARING PROPSHAFT BEARINGS
IDENTICAL WITH XTB-38 GEAR BOX.

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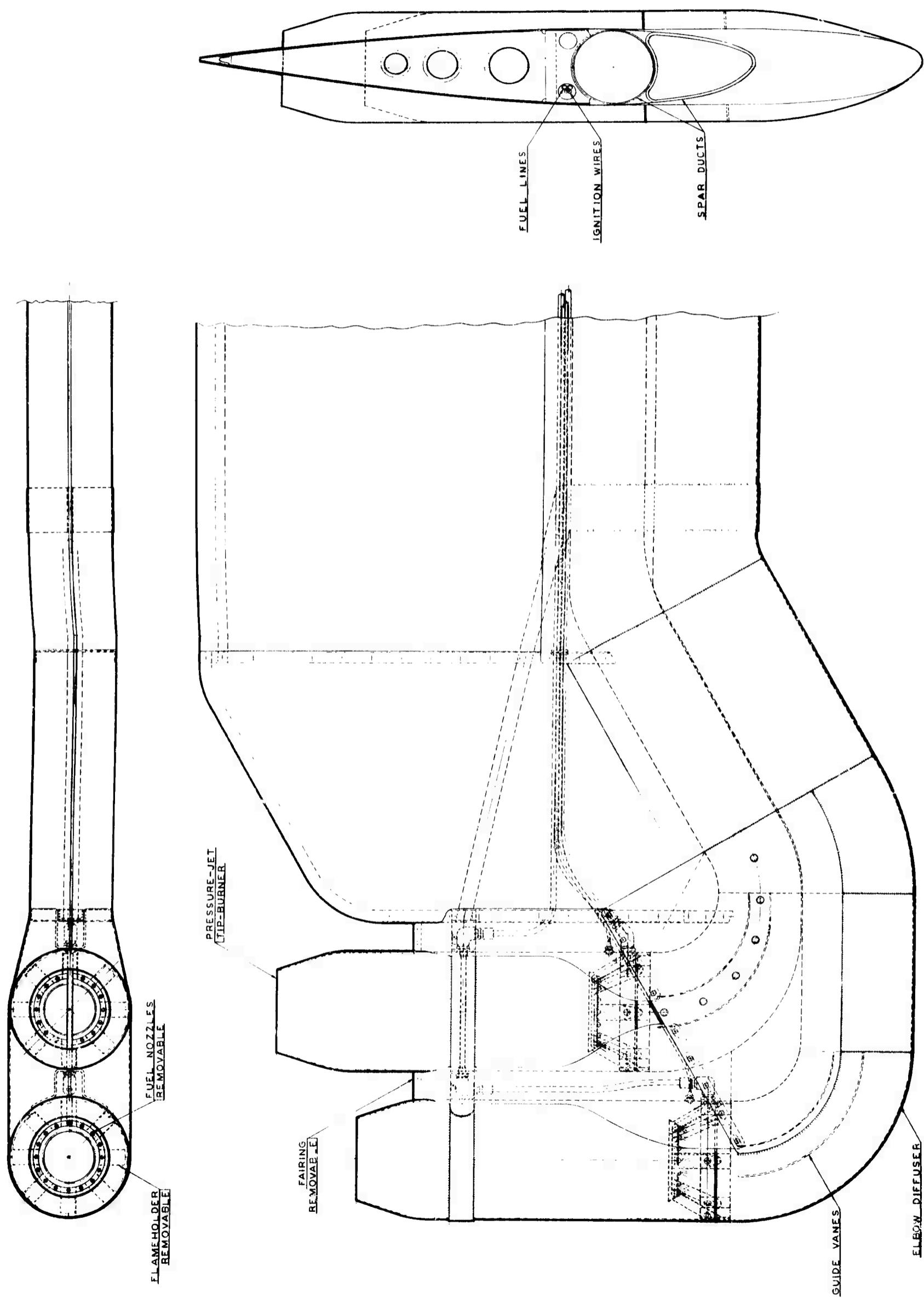
clutch which disengages the compressor. The rear face of the gear box provides the necessary accessory mounting pads and gear drives for the engine starter, generator, hydraulic pump, tachometer, and the connections for the gear box oil system and the propeller brake and governor controls. The starter gear train is connected to the engine drive shaft on the engine side of the clutch permitting the engine to be started while disengaged from the gear box.

1.6 Engine Starter - A single Ai research air bleed gas turbine compressor is employed to drive a pneumatic starter mounted on each of the two engine gear boxes.

1.7 Rotor Tip Pressure Jets - Two burners are mounted at the tip of each of the three rotor blades. Compressed air supplied from the compressor is delivered to the tip burners through a duct system. Fuel is injected into the tip burners where combustion occurs, initiated by spark ignition.

1.8 Fuel System - The Model 78 fuel system is shown schematically in figure 1. This system was designed in accordance with specification SR-73D. Fuel cells totalling 627 gallons may be filled in a conventional method through the filler necks provided in each wing or by a single point pressure fueling fitting located on the left side of the fuselage. Refueling is made possible through the pressure fueling system by opening a shut-off valve located in the fuselage. This valve is normally closed to prevent inter-cell fuel flow through the pressure fueling system, while check valves prevent outboard fuel flow from the fuselage tank.

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Fuel is delivered to the engine fuel control by two submerged boost pumps located in the sump of the fuselage fuel cell. The fuel to each power section enters through a common selector valve. Tip turner fuel flow is controlled by a rotor driven governor-pump unit. Fuel inlet pressure for this unit is provided by the engine boost pump. When the tip burners are inoperative, fuel to the rotor system is shut off by a valve internal to the rotor governor. For single engine operation, a solenoid valve, located in the rotor hub shuts off the fuel to half of the tip burners.

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MODEL 44

2. DESCRIPTION OF AIRCRAFT

The aircraft is a single place, single engine, low wing monoplane. It is built entirely of aluminum, except for the tail surfaces which are made of wood. It is powered by a Pratt & Whitney R-985-11 engine, which develops 450 horsepower at 2,200 rpm. The engine is mounted in the rear of the fuselage, driving a three-bladed propeller. The fuselage is a semi-monocoque construction, made of aluminum alloy. The wings are made of wood, with a fabric covering. The landing gear is a fixed, tail-dragger type, with a single shock absorber. The tail is a conventional tail, with a single vertical fin and rudder. The aircraft has a maximum speed of 220 mph at 12,000 ft, and a range of 400 miles. The aircraft is controlled by a stick and rudder system. The engine is controlled by a carburetor and a supercharger. The aircraft has a maximum weight of 2,500 lbs. and a maximum fuel capacity of 100 gallons. The aircraft is also equipped with a radio compass and a gyroscopic compass.

It is believed that the aircraft is a good candidate for a fighter plane, due to its speed and maneuverability. It is also believed that the aircraft could be used for transport purposes, due to its spacious interior and ability to carry a payload of up to 500 lbs.

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MODEL 15

MODEL _____

During the next few days the weather was very bad.

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operating at the lower speed.

In order to obtain optimum pressure jet performance during single engine operation when the air flow is only one half of the normal flow, it is necessary to reduce the tip burner total exhaust nozzle area by 50 per cent. This is accomplished by closing the butterfly valves located in the aft duct in the root of each rotor blade, thus removing the three inner tip burners from the system. Actuation of this butterfly valve is automatically controlled as explained in section 2.3.

2.3 Pressure Jet Fuel Control - Fuel flow to the rotor tip burners is controlled by a rotor speed governor which meters fuel as necessary to maintain a constant selected rotor speed during all jet powered rotor operation. Thus as the rotor load changes, the pressure jet fuel flow is automatically adjusted to hold rotor speed, completely relieving the pilot of this duty. This type of rotor speed and tip jet fuel control has been performing satisfactorily in flight for a period of three years on the McDonnell XH-20 ram jet powered helicopter.

During periods of single engine operation, the flapper door of the air control valve (see section 2.2) actuates switches which closes a fuel shut-off solenoid valve located in the rotor hub. This solenoid valve shuts off the fuel flow to the three inner tip burners. Also connected to the manifold supplying fuel to the inner set of tip burners are actuating cylinders controlling the butterfly valves located in each rotor blade in the ducts supplying compressed air to the inner set of tip burners (see section 2.2). These cylinders are arranged so as to hold the butterfly valves open when fuel pressure exists in the manifold supplying fuel

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to the inner set of tip burners, and allows the butterfly valves to close when the solenoid valve shuts off the fuel in this manifold.

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 MODEL 7B

SYMBOLS

A	area	ft ² unless otherwise noted
a	speed of sound	ft/sec
c _p	specific heat at constant pressure	BTU/lb°R
D	hydraulic diameter	ft
F	thrust	#
F ₀	compressibility factor	$1 - \frac{M^2}{4} + \frac{M^4}{40} - \frac{M^6}{1600} + \dots$
f	wall friction factor	
f/a	fuel-air ratio	
g	acceleration of gravity 32.2 ft/sec ²	
H	total head	#/ft ²
HP	horsepower	
H ₂ /E ₀	total pressure recovery	
h _v	lower heating value 18,000 BTU/lb for gasoline	
J	mechanical equivalent of heat 778 ft-lb/BTU	
l	length	ft
M	Mach number	
m	mass rate of flow	slugs/sec
P	absolute static pressure	#/ft ² unless otherwise noted
P _T	absolute total pressure	#/ft ² unless otherwise noted
Q	volume rate of flow	#/ft ² unless otherwise noted
q	dynamic pressure	#/ft ² unless otherwise noted
q ₀	impact pressure qF ₀	#/ft ² unless otherwise noted
R	gas constant	ft-lb/#/R

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RN	Reynolds number	
S.H.P.	shaft horsepower	
T	static temperature	°R
T _T	total temperature	°R
V	velocity	ft/sec
V _T	tangential velocity	ft/sec
v	specific volume	ft ³ /lb
W	weight rate of flow	lb/sec
α	angle of attack	degrees
γ	ratio of specific heats	
Δ	used to represent a change in another quantity	
δ	$P_{T_2}/14.7$ when H_{T_2} is expressed in PSIA	
θ	$T_{T_2}/18.4$	
η	efficiency	
ρ	mass density	slugs/ft ³
ω	rotational velocity	radians/sec

Subscripts

0	free stream	i	inlet
1, 2, etc.	station numbers	j	jet
a	air	ot	total loss
b	burning	p	primary
c	compressor	s	secondary
f	fuel	TR	temperature rise
g	gas - products of combustion		

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3. ANALYSIS OF POWER PLANT DUCTS

3.1 General - The analysis of the Model 78 duct system is presented in accordance with the U. S. Navy Aeronautical Specification NAVAR 5B-180, "Specification for the Calculation of Duct losses, Turbo-Jet and Gas Turbine Engines".

This section includes the analysis of the inlet ducts supplying air to the two Allison Model 501 power sections, the inlet ducts supplying air to the two Westinghouse 19XB compressors, and the engine exhaust system including the cooling ejector.

3.2 Description of Ducts - The engine inlet duct is approximately five feet long with a well-rounded lip. The duct increases in cross-sectional area from 160 square inches at the inlet to 186 square inches at the engine face. The annular flow area at the engine compressor face is 160 square inches.

The 19XB compressor inlet duct is about three feet long and has a well-rounded lip. The duct increases in cross-sectional area from 140 square inches to 160 square inches at a station near the compressor face. The annular area at the compressor face is 138 square inches.

The engine exhaust tailpipe is shrouded with an ejector which is split by radial dividers into three parts drawing cooling air through the forward engine compressor compartment, the engine combustion chamber compartment, and the oil cooler.

Figure 3 presents a schematic of the ducting arrangement and cross-sectional areas.

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3.3 Results of Analysis - The total pressure recovery for the engine and the compressor inlets is shown in figure 4. It is based on a series of wind tunnel and flight tests conducted by MAC on comparable installations (referenced 1 and 2).

Tests of similar installations indicate that the inlet and compressor inlet ducts should have a total pressure recovery of 92% under sea level static conditions without sonic or supersonic flow.

Tables I and II present the detailed analysis of the inlet system - cover, inlet duct, fan and compressor. At sea level with ambient $T = 50^{\circ}\text{F}$, cross section area, equivalent to a total of 300 sq. inches, and a total velocity of 100 ft/sec, static air pressure is 14.7 psia and 1.8 psf at 8 ft above the ground surface.

The analysis will be applied to the inlet system of figure 4. This inlet system is similar to that used in figure 1 and figure 2. The inlet duct is 2.5 in. in diameter and has a length of 8 ft. The inlet is located 8 ft above the ground surface or a MAC altitude of 8,000 ft.

Data plotted in reference 1 shows that a secondary air flow of 0.3 l/sec may be induced at sea level at a height of 8,000 ft. (static pressure is normal power). This ejector is shown to be increased at 15,000 ft. for the lower altitude flying requirement. Terms of a 15,000 ft altitude correction factor and the total area of ejector at 15,000 ft. are not known. No losses may be expected from this factor. Cross section areas, equivalent conical angle of diffusion, and exhaust Mach number in an axial direction for the inlet ducts are presented in figures 11 and 12.

The analysis of the desired air duct system for the compressor to the afterburner inlet is based on the same basic assumptions, included in the pressure jet problem analysis.

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4. ALLISON MODEL 501 POWER SECTION PERFORMANCE

4.1 Engine Performance - The Allison Model 501 gas turbine engine performance was evaluated by the method presented in the engine specification (reference 3). Reference 3 represents the most recent Navy approved specification for the power section performance. Inlet duct total pressure recovery was based upon the duct analysis presented in section 3. Values of H_2/H_0 from figure 4 were used for the engine performance calculations.

During powered-rotor flight with the propeller in the pitch resulting in minimum power absorption it is assumed that the propeller absorbs 6% of the available engine shaft horsepower. Accordingly the power available to drive the compressor, as presented in figure 13 has been reduced by 6%.

As noted in section 2.1 the engine will be operated at constant normal speed (14,000 rpm) during all periods that it is engaged to the compressor. This necessitates, under certain conditions, that the turbine inlet temperature exceed the normal rated temperature, but under no conditions will it exceed the allowable military temperature. In view of the allowable operating time of 30 minutes at military power this type of operation is considered satisfactory especially as it is accomplished at a reduced engine speed. Figure 13 presents a plot of horsepower available and horsepower required versus altitude for the most critical conditions at 14,000 rpm and a turbine-in temperature of 19.5°R .

Figures 14 through 22 present shaft horsepower, net jet thrust, and fuel flow for propeller operation of each engine in the anticipated operating ranges of flight velocity and altitude. Performance was determined in accordance with the engine specification (reference 3) using the estimated inlet total pressure recovery presented in figure 4.

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MODEL 10

• 1.1 Overall Airflow Performance

1.1.1 Overall Airflow Performance - The overall performance data of this engine is determined by the following information listed in the text which follows itself. If the required overall performance is not obtained it was necessary to construct a new horsepower versus pressure ratio curve based upon the available temperature rise efficiency data available in reference 6. The equation:

$$HP = \frac{1}{n_{TR}} \frac{RT}{550} \left(\frac{\gamma}{\gamma-1} \right) \left[\left(\frac{P_3}{P_{T_2}} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right] Wa \quad 1$$

was employed for this purpose. The resulting curve (Figure 2) was checked by Neumann and found to be admissible for all purposes. The curves required prior to calculation are Figure 2a and 2b are represented in reference 6.

1.1.2 Tip Burner Performance - The tip burner performance is determined from the relationship of the primary air flow rate and the air entering the combustion chamber. The primary air and air entering will be denoted by P_1 and P_2 respectively. The primary air enters the combustion zone at a rate of W_a . A graph of this relationship is shown in Figure 2a.

1.1.3 Low Pressure Turbine Air Flow - The air flow through the low pressure turbine is determined from the following equation:

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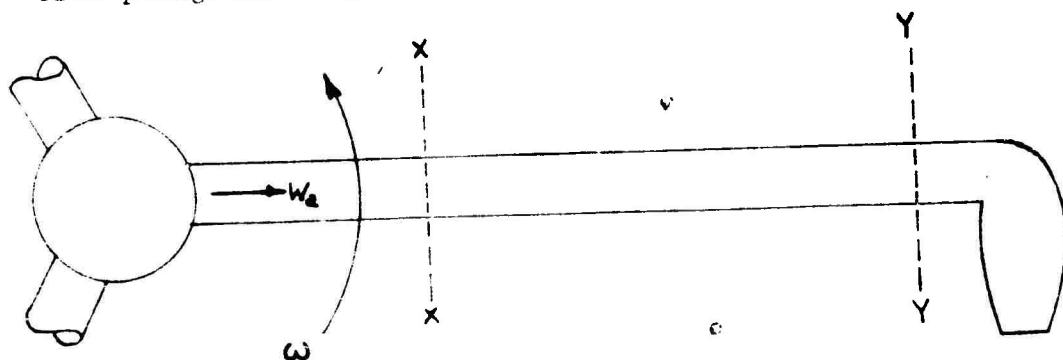
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Considering a rotor blade as shown below with air flowing through the blade passage and being burned in the tip burner, the power required to



accelerate the air from the tangential velocity at station X to the tangential velocity at station Y is:

$$\text{POWER} = \frac{W_a (V_{T_y}^2 - V_{T_x}^2)}{2g}$$
2

also, the power required for isentropic compression of the air from station X to station Y is: (reference 7).

$$\text{POWER} = J C_p T_{T_x} \left[\left(\frac{P_{T_y}}{P_{T_x}} \right)^{\frac{k-1}{k}} - 1 \right] W_a$$
3

Equating equation 2 and equation 3

$$\frac{W_a (V_{T_y}^2 - V_{T_x}^2)}{2g} = J C_p T_{T_x} \left[\left(\frac{P_{T_y}}{P_{T_x}} \right)^{\frac{k-1}{k}} - 1 \right] W_a$$

$$\left(\frac{P_{T_y}}{P_{T_x}} \right)^{\frac{k-1}{k}} = 1 + \frac{(V_{T_y}^2 - V_{T_x}^2)}{2g J C_p T_{T_x}}$$

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$$\frac{P_{T_y}}{P_{T_x}} \left[1 + \frac{(V_{T_y}^2 - V_{T_x}^2)}{2gJ C_p T_x} \right]^{\frac{g}{g-1}}$$

Since $JC_1 = R(\gamma/\gamma-1)$, the equation for the pressure ratio developed from station X to station Y may now be written:

$$\frac{P_{T_y}}{P_{T_x}} = \left[1 + \frac{(V_{T_y}^2 - V_{T_x}^2)}{2gR T_x (\frac{g}{\gamma-1})} \right]^{\frac{g}{g-1}}$$

4

2.2.2 Pressure Losses - To determine the pressure losses in the system due to the flow of air out of the ducting to the tip burner, a step-by-step analysis was made for each calculated tip burner geometry. The method and data of reference 8 were used to evaluate the pressure losses. Pumping gains due to rotor rotation, including the pressure profile discussed in section 2.1 and momentum pressure loss (calculated at reference 7) were also taken into consideration. After the pressure available in the air plenum for combustion was determined, the tip burner performance was evaluated in the manner discussed below.

2.2.3 Design Condition - A design condition was first selected which was the basis for determining the nozzle exit area of the tip burner. The design condition chosen was a free stream pressure field (\bar{P}/\bar{P}_0) of 17,000

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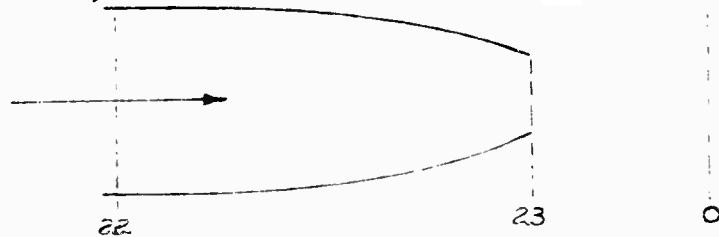
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MODEL - 1P

operating at a certain pressure ratio ($\frac{P_{23}}{P_{22}}$) the nozzle is at a critical point of the nozzle where efficiency is maximum. This pressure ratio is sufficient below the compressor train to indicate a safe condition. Since the nozzle has a safety factor of 1.5 at this design condition indicates the pressure is sufficient to maintain sonic velocity through the nozzle exit area, the area which is selected on this basis.



$$P_{cr} = RT$$

$$\rho = \frac{RT}{P} = \frac{AV}{W_g}$$

$$\frac{W_g RT_{22}}{P_{23}} = A_{23} V_{23}$$

For a given exit area, the critical exit

$$V_{23} = \sqrt{\frac{W_g RT_{22}}{P_{23}}}$$

From critical pressure ratio it is possible to write

$$\frac{P_{23}}{P_{T_{22}}} = \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$$

$$P_{23} = P_{T_{22}} \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$$

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Substituting equations 6 and 7 into equation 5 the following expression is obtained for the nozzle area.

$$\frac{W_g R T_{23}}{P_{T_{22}} \left(\frac{2}{8+1}\right)^{\frac{8-1}{8+1}}} = A_{23} \sqrt{8g R T_{23}}$$
8

$$A_{23} = \frac{W_g R T_{23}}{P_{T_{22}} \left(\frac{2}{8+1}\right)^{\frac{8-1}{8+1}} \sqrt{8g R T_{23}}}$$
9

In view of the need for maximum burning efficiency it is apparent that no performance improvement of 4,000' banking will be obtained in the combustion chamber. Test data, as a similar one referred to in reference 9 indicates corner to corner is of this order and higher. Therefore the nozzle exit area was determined for the maximum air flow coefficient at a total temperature of 4,000° R. Internal cooler layer air cooling is used in the wall. The condition is presented in the following table:

Internal cooler layer air cooling is used in the wall. The condition is presented in the following table:

$$T_T = T \left(1 + \frac{8-1}{2} M^2\right)$$
10

SINCE $M = 1.0$

$$T_{23} = 4000 \left(\frac{2}{8+1}\right)$$

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MODEL 78

From equation 9, the nozzle area now becomes:

$$A_{z3} = \frac{W_0 R(4000) \left(\frac{2}{\gamma+1}\right)}{P_{T22} \left(\frac{2}{\gamma+1}\right)^{\gamma-1} \sqrt{\gamma g R(4000) \left(\frac{2}{\gamma+1}\right)}} \quad 11$$

$$\frac{f}{d} = \frac{C_p \Delta T}{h_v \eta_b}$$

Assume a burning efficiency of 90% and a lower heating value of gasoline
 $\approx 18,000 \text{ BTU/lb}$.

$$\frac{f}{d} = \frac{C_p \Delta T}{16,200} \quad 12$$

Since the values of γ and C_p are dependent upon temperature and fuel-air ratio it is now necessary to make successive approximations for fuel-air ratio and cycle through equations 11 and 12 until both equations are satisfied.

P_{T22} is determined from the duct analysis as described in section 5.2.2 and a trial and error solution is readily made since there is only one fuel-air ratio that will give a total temperature of 4000°R.

On this basis a total nozzle area of 0.938 square feet was determined. This total nozzle area is divided by the number of tip burners to obtain the area per burner. For the six burners the effective nozzle exit area per burner is 22.6 in.² or 5.36 in. in diameter.

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MODEL 78

5.2.4 Performance Calculations - For other corrected compressor speeds and pressure ratios the combustion temperature may be determined since the area has been determined. Transposing equation 8:

$$\frac{T_{23}}{\sqrt{T_{23}}} = \frac{A_{23} \sqrt{8\gamma R} P_{T_{22}} \left(\frac{\gamma}{\gamma-1}\right)^{\gamma-1}}{W_g R}$$

$$T_{23} = \left[\frac{A_{23} \sqrt{8\gamma R} P_{T_{22}} \left(\frac{\gamma}{\gamma-1}\right)^{\gamma-1}}{W_g R} \right]^2 \quad 13$$

Using equation 12 and 13 and determining $P_{T_{22}}$ from the rotor duct analysis T_{23} may be determined for any compressor condition. Although equation 13 applies only when $P_{T_{23}}$ is above the critical pressure ratio, this condition covers most of the system operating range.

Fuel flow is determined by:

$$W_f = f/a W_a = \frac{C_p \Delta T}{16,200} W_a \quad 14$$

Total temperature at the nozzle exit for sonic velocity is

$$T_{T_{23}} = T_{23} \left(\frac{\gamma+1}{\gamma} \right) \quad 15$$

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The jet velocity is obtained from equation 6. Jet thrust is obtained as follows:

$$F_J = \eta_N \frac{W_0}{g} (V_{23} - V_T) + (P_{23} - P_0) A_{23} \quad 16$$

η_N = nozzle efficiency (assumed .95)

A sample calculation is presented in table 7. The first section of the calculation is the rotor system duct analysis to determine the pressure available for burning. The latter part of the calculation presents the tip burner performance for the available pressure. It was assumed that the division of air flow through the two flow passages in the rotor blade (station 17 through 23) was equal, even though the cross-sectional areas are slightly different (22.1 and 24.3 square inches per blade). The actual division of the air flow will be governed by the back pressure of each tip burner due to burning. The total temperature, fuel flow, and jet thrust were calculated for each of the two tip burners and the average total temperature was used for the total temperature of the gas.

Curves of corrected performance over a range of thrust values are presented in figures 26 and 27. Corrected values are presented in the general curves since they are independent of the compressor inlet conditions. The correction factors used are the standard factors used in correction of jet engine performance. Reference 10 presents the derivation of these factors. These factors are:

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$$\delta = P_{T_2} / 14.7 \text{ where } P_{T_2} \text{ is expressed in PSIA}$$

$$\Theta = T_{T_2} / 518.4 \text{ where } T_{T_2} \text{ is expressed in degrees Rankine}$$

Several numerical examples were checked for various altitude, within the performance range of Model 78. Agreement of corrected performance with that determined by using the actual temperature and pressure was obtained.

Actual tip burner thrust and overall fuel consumption are presented in figures 28 through 31 for the anticipated operating range of the tip burners.

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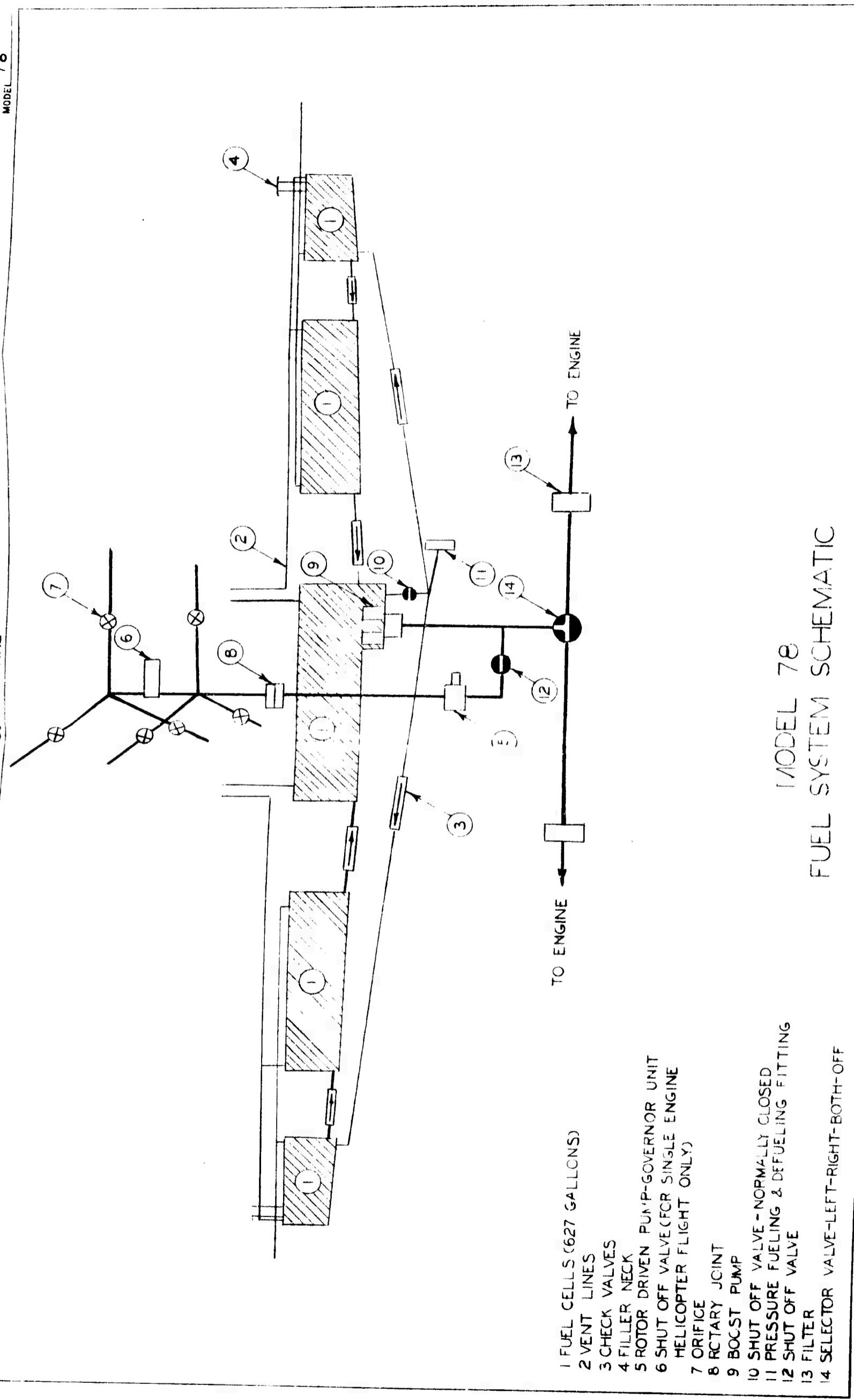
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MODEL 78
FUEL SYSTEM SCHEMATIC

FIG 1

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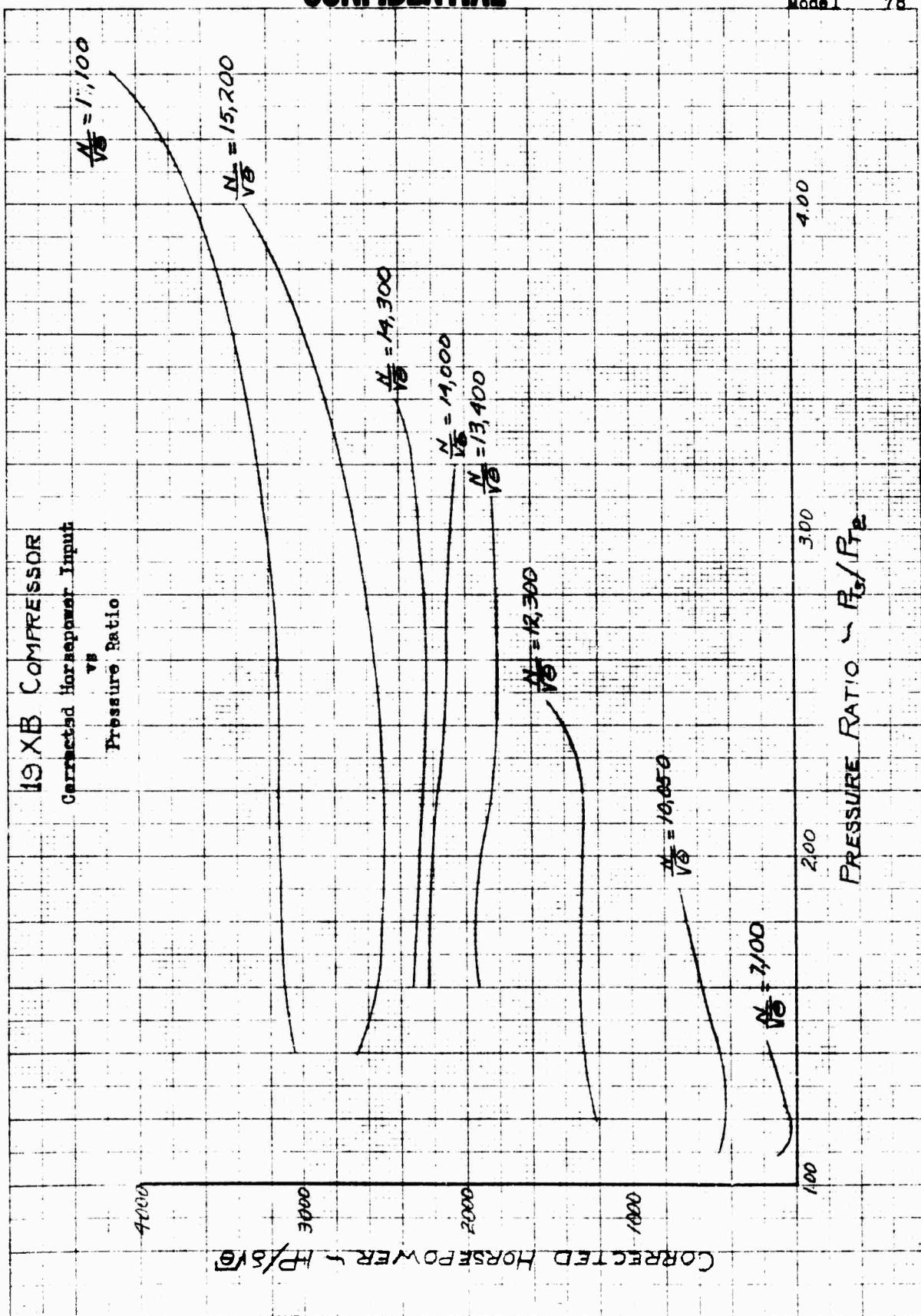


FIG. 2

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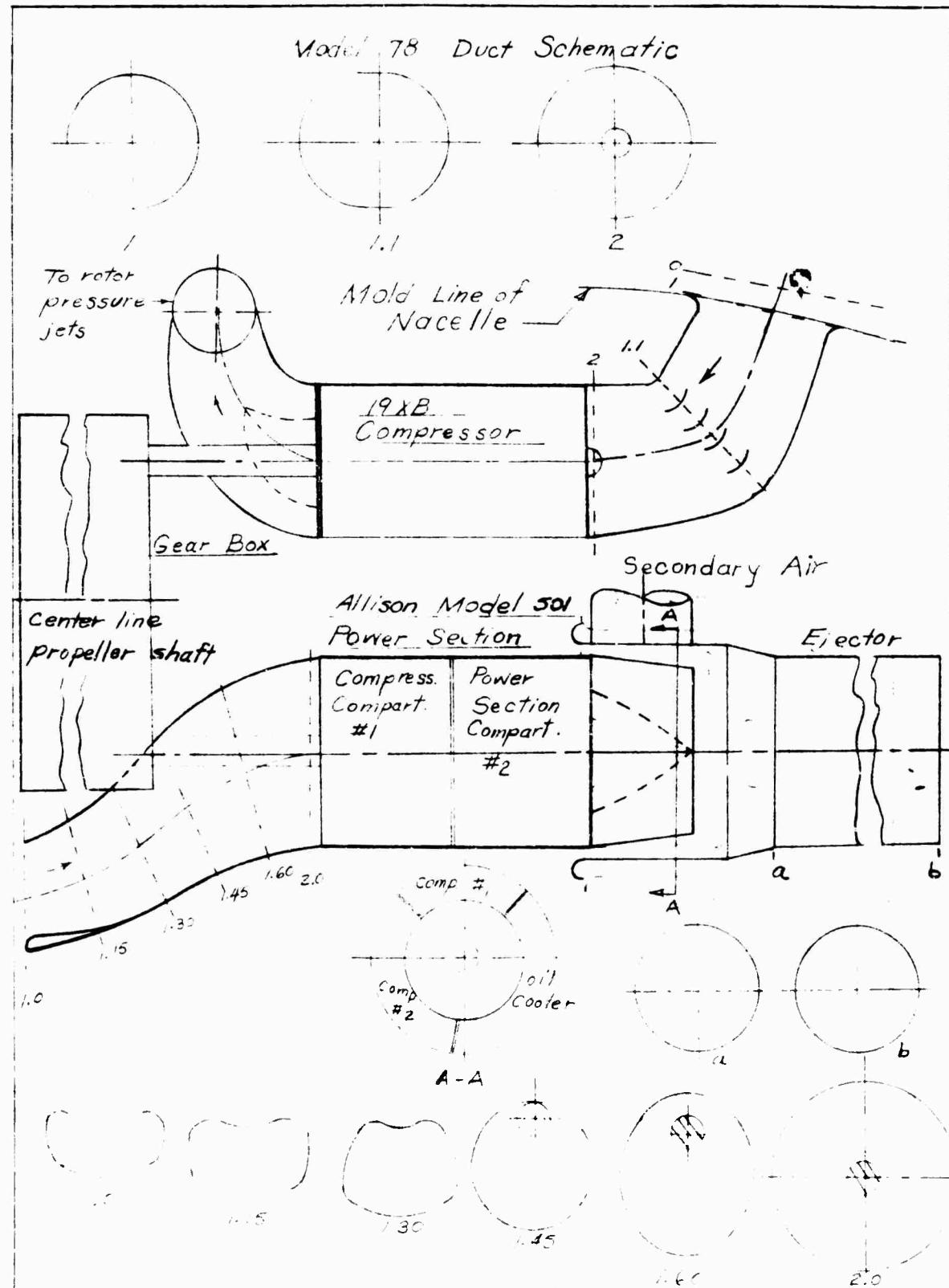
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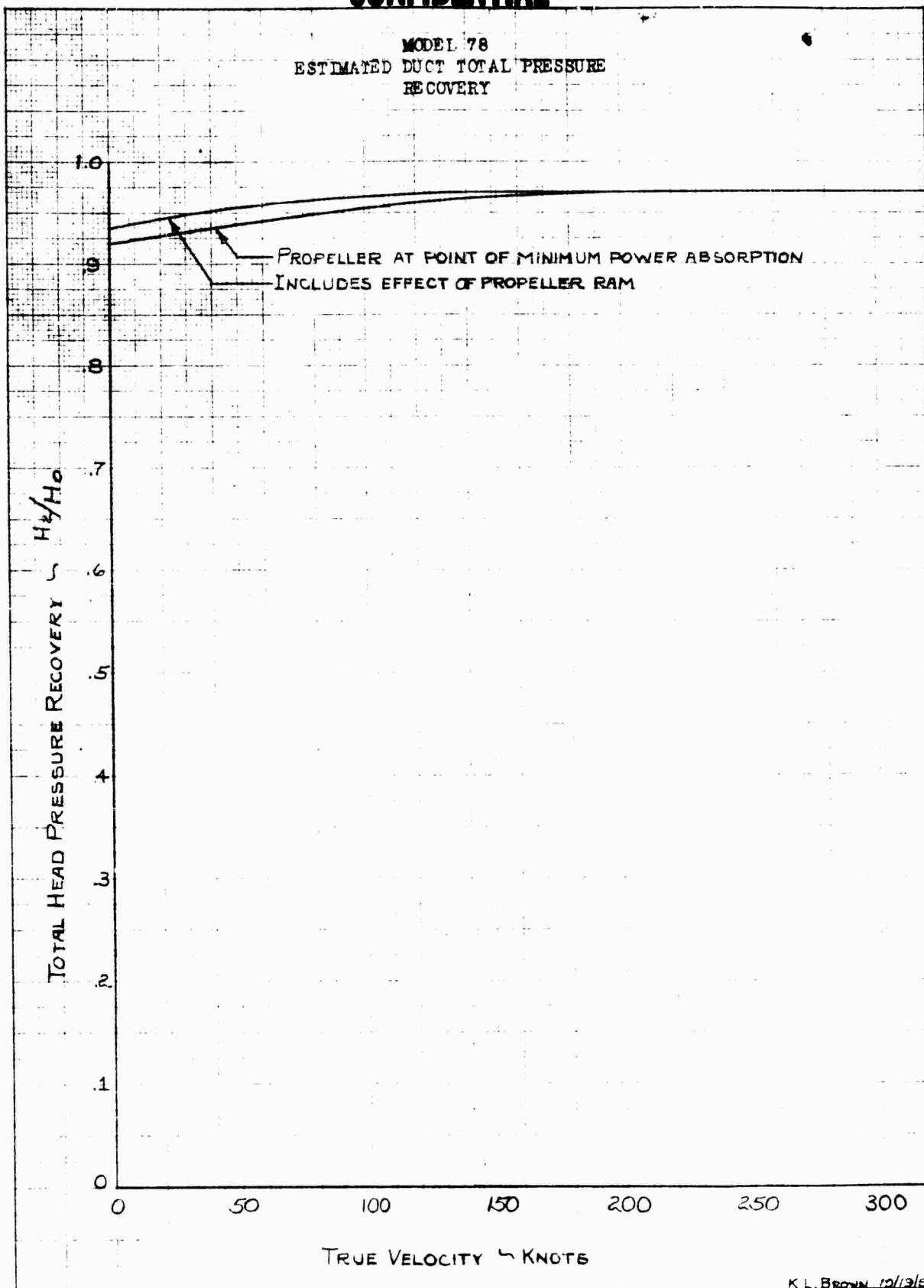
FIG. 3

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MODEL 78
ESTIMATED DUCT TOTAL PRESSURE
RECOVERY



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FIG. 4

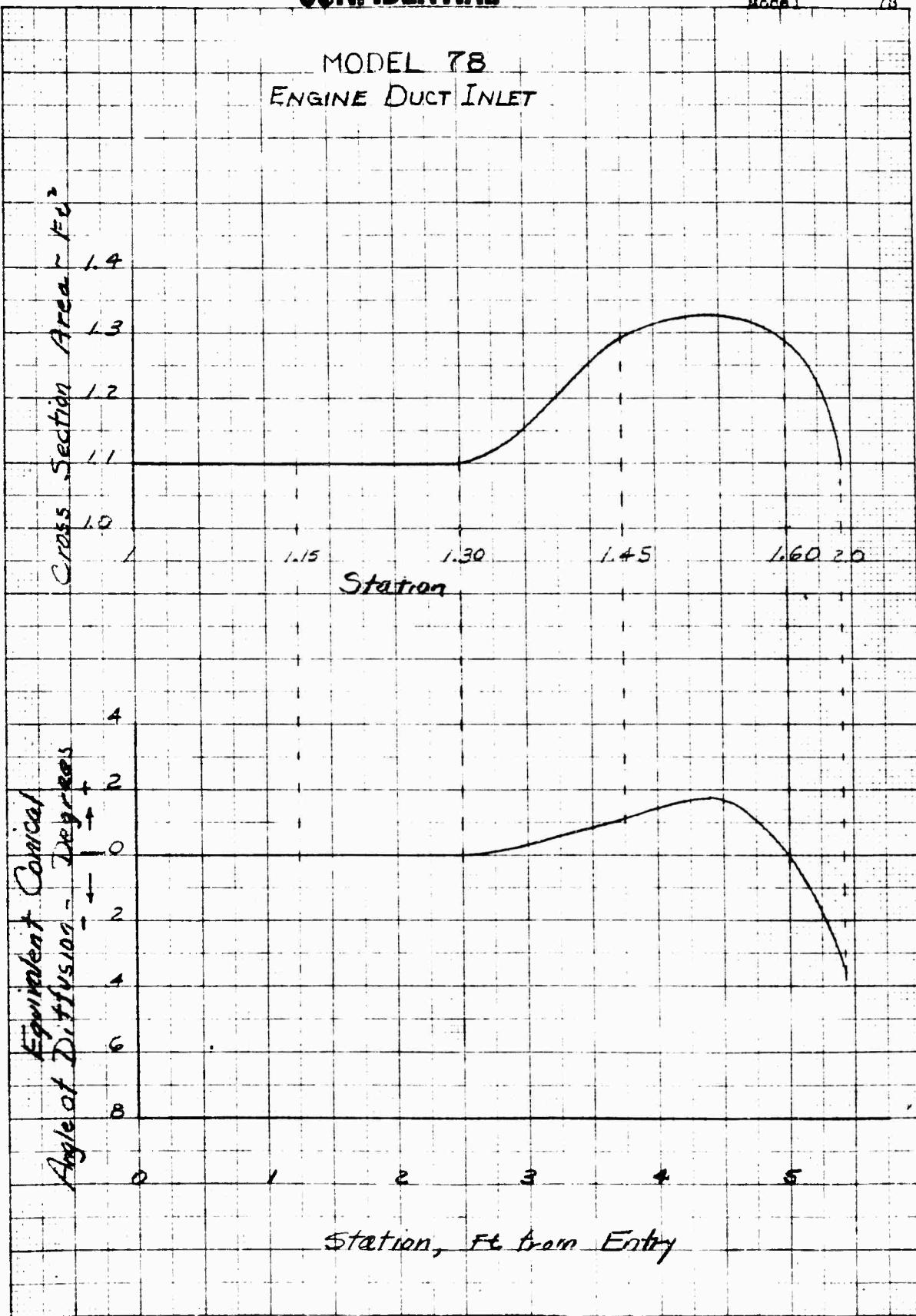
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MODEL 78
ENGINE DUCT INLET



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FIG. 5

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Model 78
 Engine Duct Inlet
 Sea Level, Static
 Normal Power

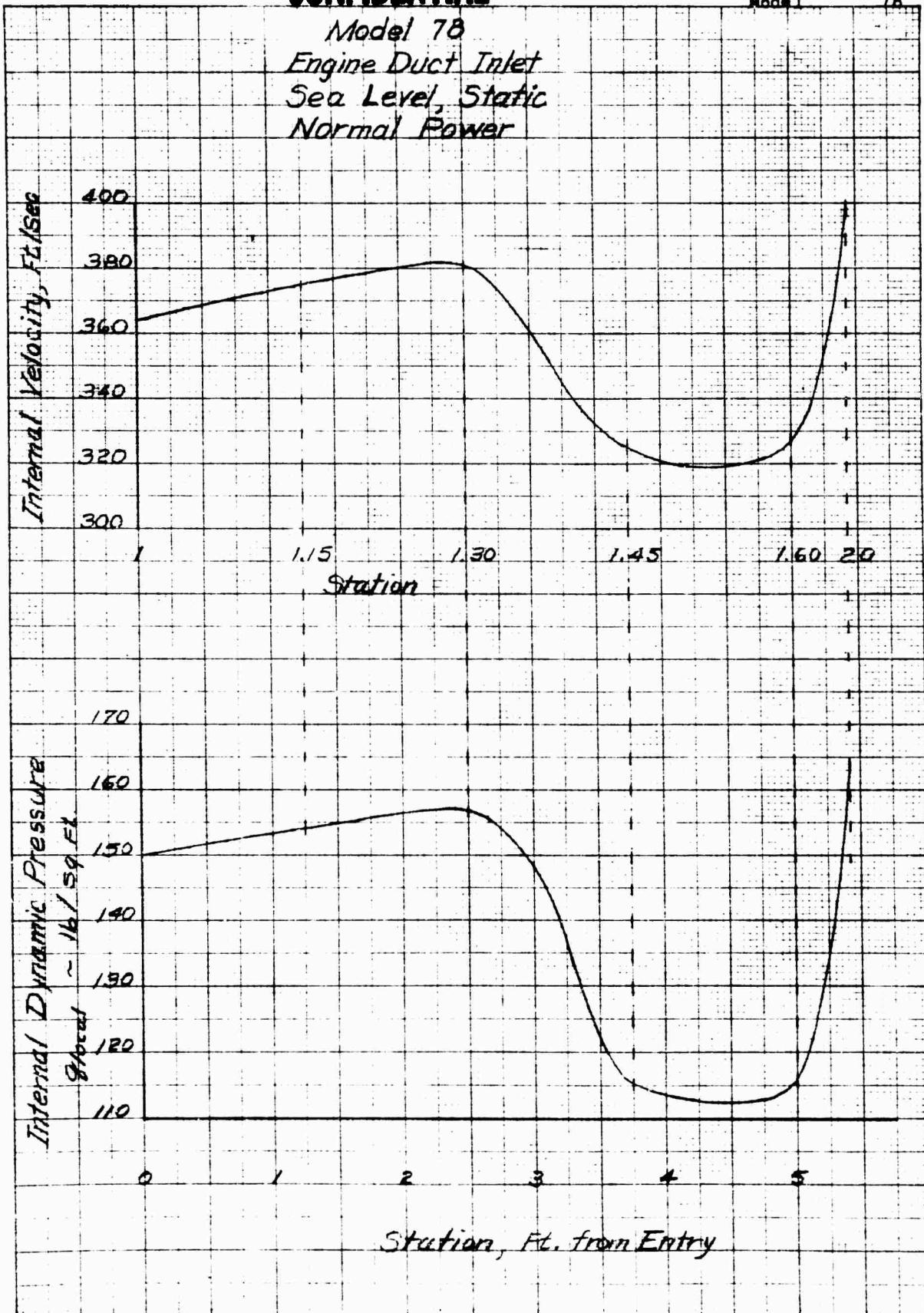
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FIG. 6

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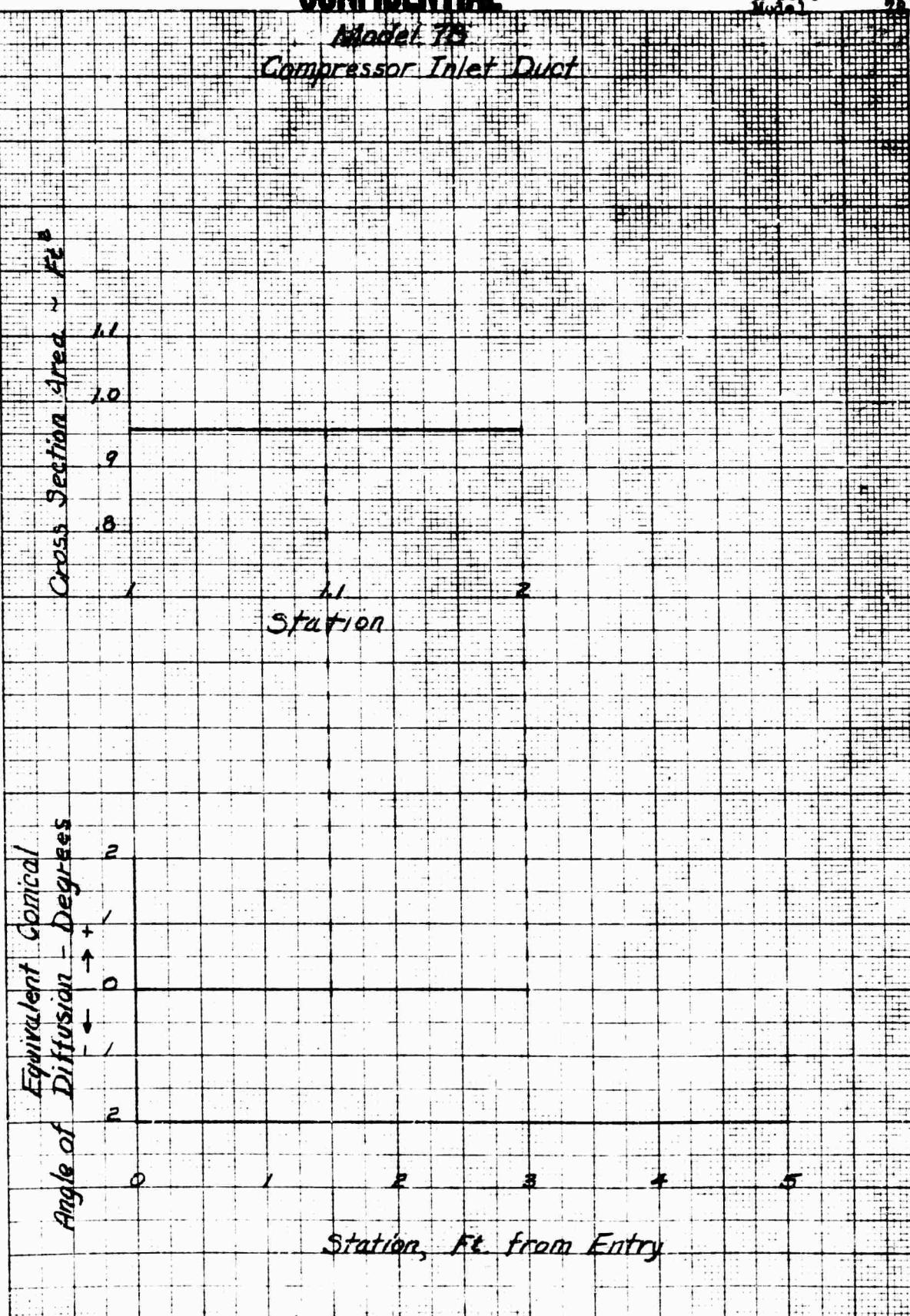
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Model 7B

Compressor Inlet Duct



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FIG. 7

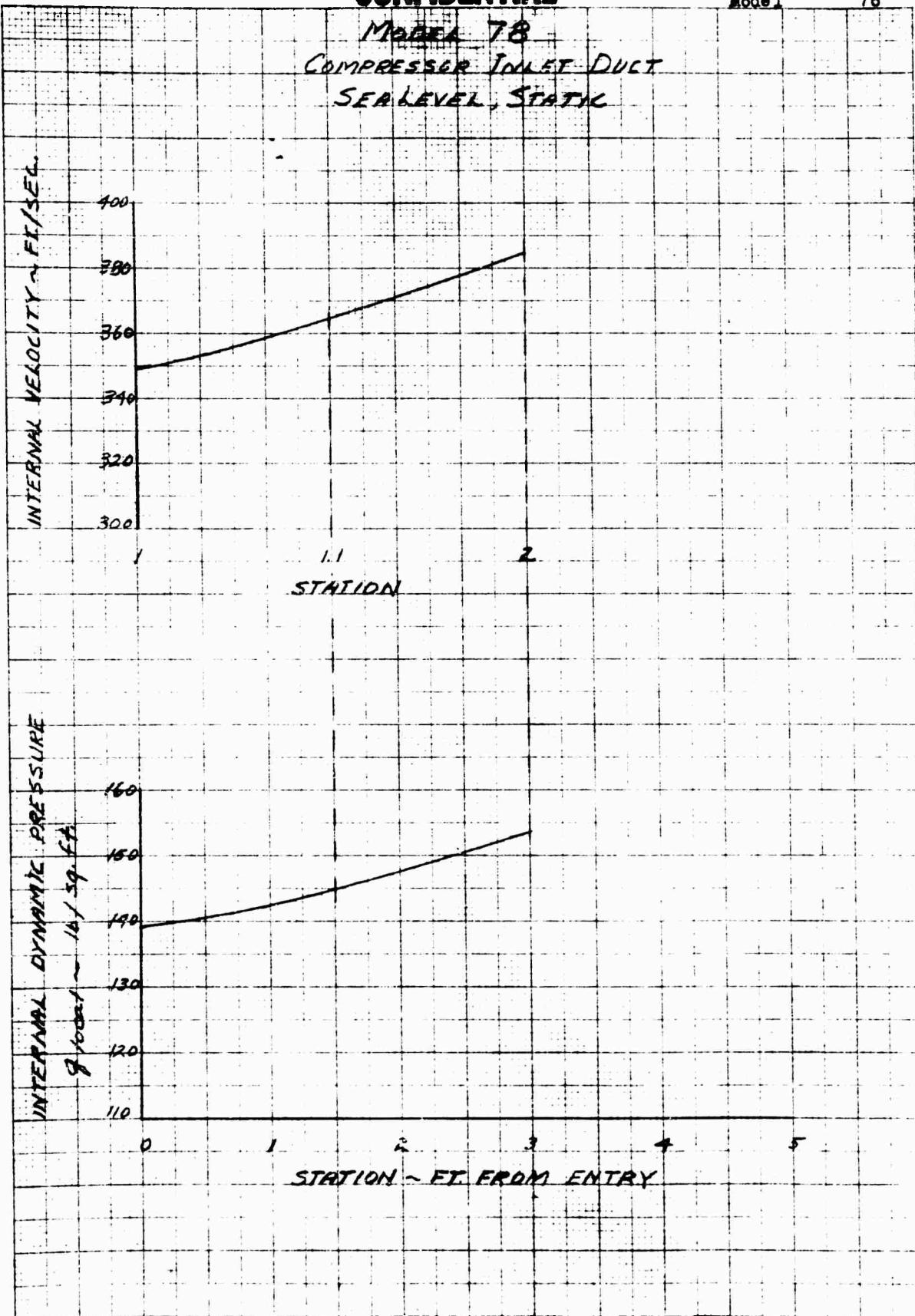
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MOTOR 78
COMPRESSOR INLET DUCT
SEA LEVEL, STATIC



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FIG. 8

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Condition: Sea Level, Static, Normal Power (4,500 RPM)

$$P_0 = 573 \text{ #} \quad \text{Revolving Pressure, } T_0 = 75^\circ\text{F}$$

$$W_0 = 28.45 \text{ #/sec}, \quad W_T = 14.73 \text{ #/hr} = .002 \text{ #/sec}$$

$$W_T/W_0 = .0144, \quad M = 1.353, \quad T_0 = 575^\circ\text{R}$$

$$F = \frac{P}{T} \left(\frac{P_0}{P} \right) \frac{T_0}{T} = \frac{W_T}{W_0} = \frac{.002 \times 23.45 + .002}{.002} \times 75 = 0 = 575$$

$$V = \frac{575 \times 575}{.75 (28.45)} = 676 \text{ Ft/sec}$$

From the ideal jet velocity equation, solve for the nozzle pressure ratio.

$$\frac{P}{P_0} = \sqrt{\frac{P_0}{P}} \left[1 - \left(\frac{P_0}{P} \right)^{\frac{2}{k-1}} \right]$$

$$\frac{P}{P_0} = \frac{2 \times 22.2 \times 53.9 \times 1.353 \times 1295}{1.353} \left[1 - \left(\frac{P_0}{P} \right)^{\frac{2 \times 1.353}{1.353-1}} \right]$$

$$\frac{P}{P_0} = \left(\frac{P}{P_0} \right)^{\frac{3.85}{2}} = .90$$

$$\frac{P}{P_0} = \frac{1}{.90} = 1.12$$

The exhaust nozzle pressure ratio is 1.12.

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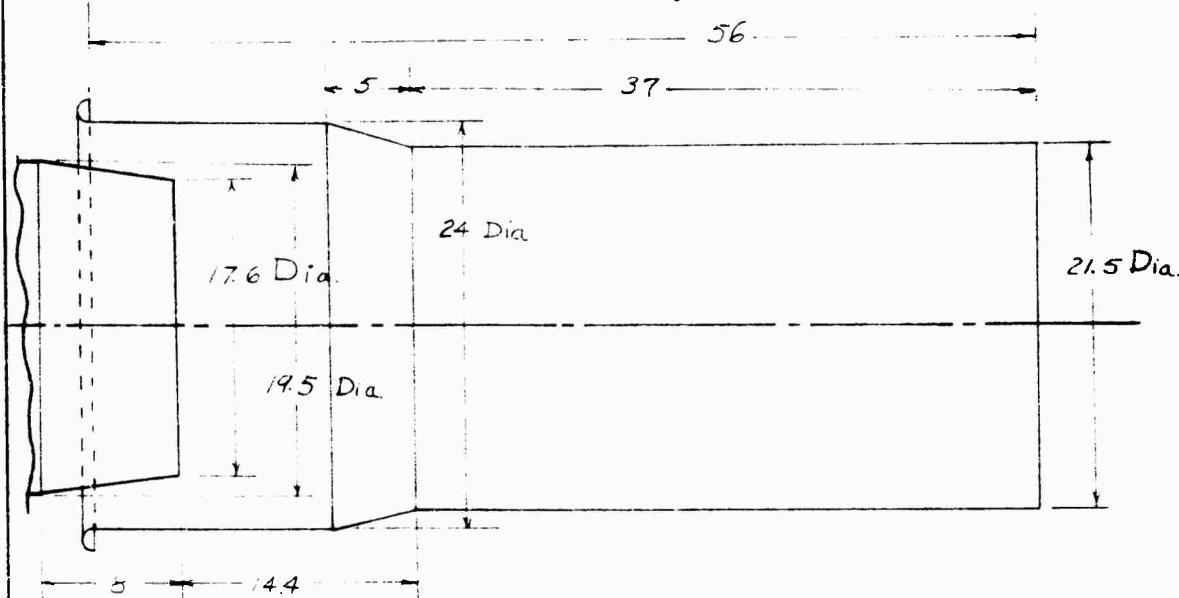
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Estimated Secondary Weight Flow*The corrected weight flow ratio from reference 4.*

$$\frac{w_s}{w_p} \sqrt{\frac{T_s}{T_p}} = .14$$

$$T_s = 520^{\circ}R$$

$$T_p = 1295^{\circ}R$$

$$w_s/w_p = .14 \sqrt{\frac{T_p}{T_s}} = .14 \sqrt{\frac{1295}{520}} = .222$$

$$w_p = 28.45 \text{ lb/sec}$$

$$w_s = .222 \times 28.45 = 6.28 \text{ lb/sec}$$

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FIG. 10

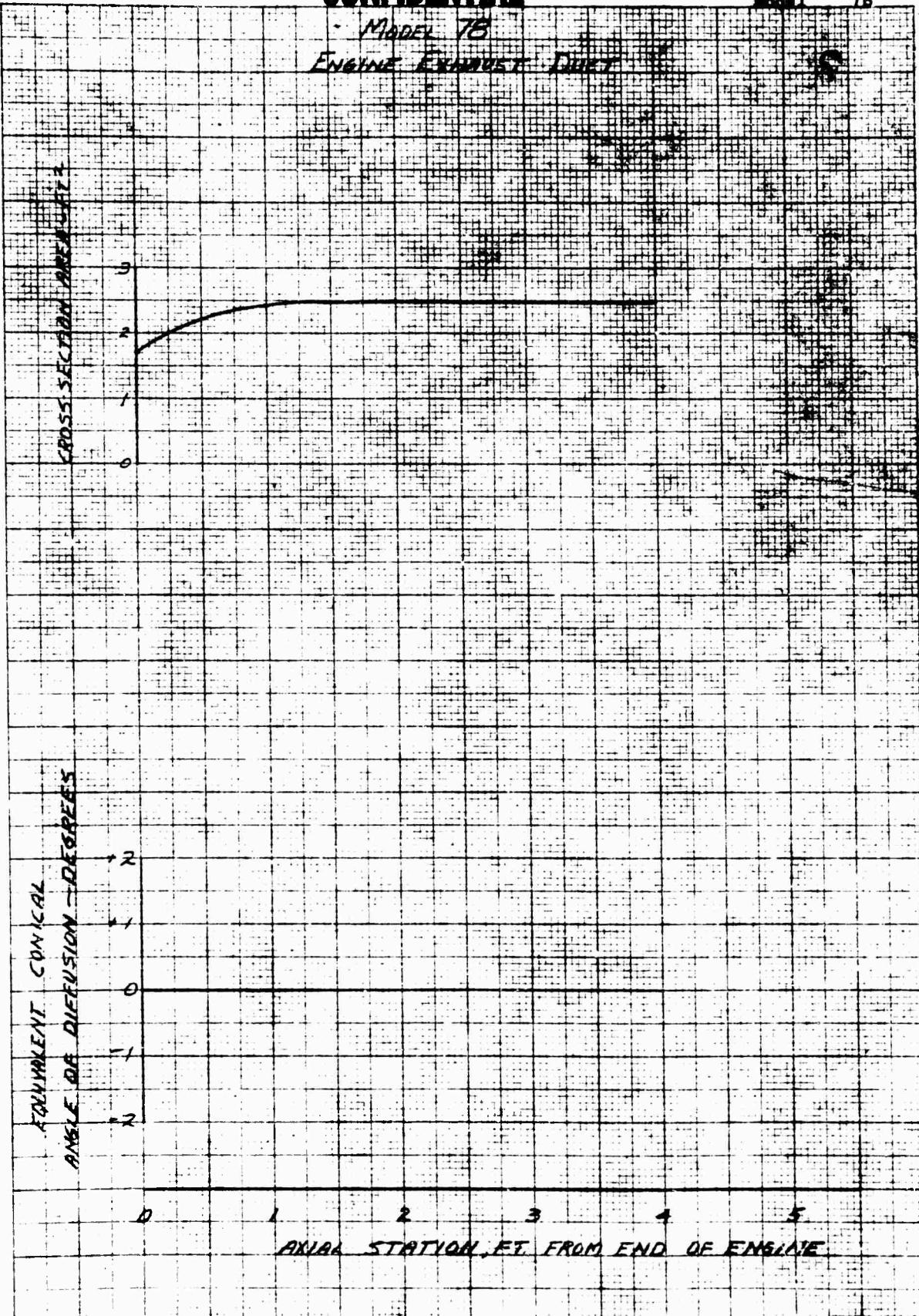
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- MODEL 78
ENGINE EXHAUST DRAFT

FIGURE 11. INFLUENCE OF DIFFUSION ANGLES ON DRAFT.
PROJECTION OF DRAFT ON A PLANE PERPENDICULAR TO THE DRAFT.



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FIG. 11

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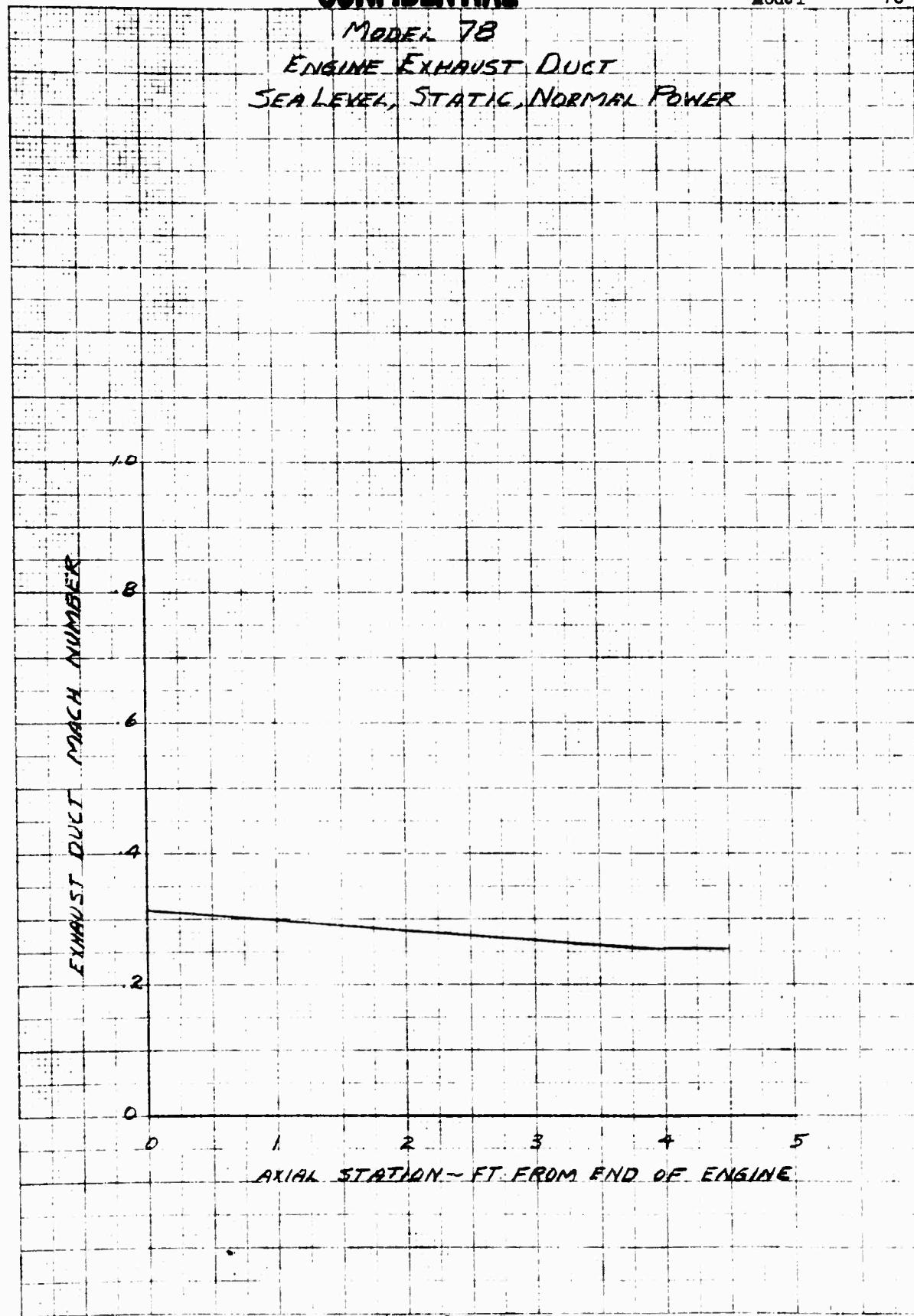
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MODEL 78

ENGINE EXHAUST DUCT
SEA LEVEL, STATIC, NORMAL POWER

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FIG 12

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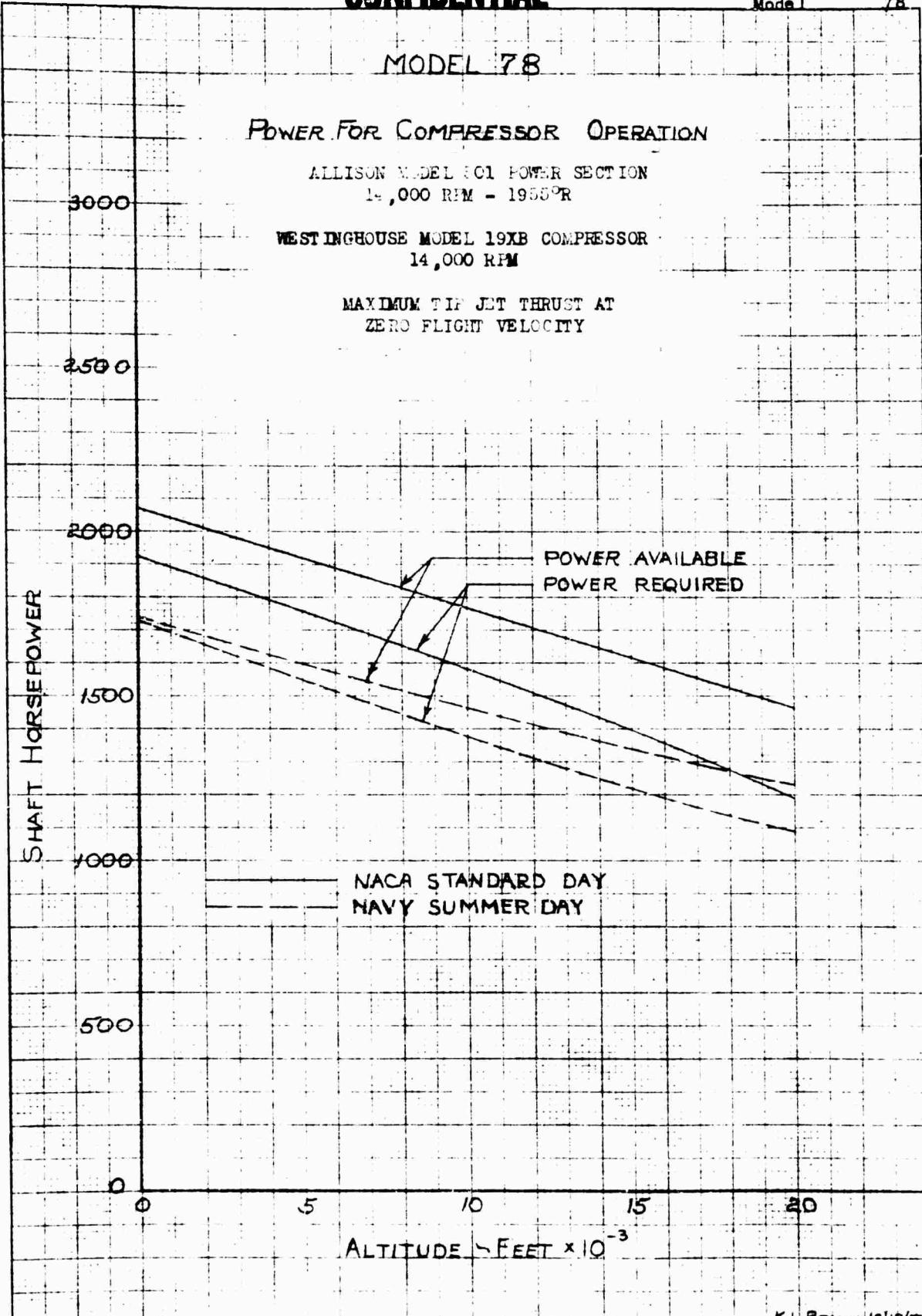
POWER FOR COMPRESSOR OPERATION

ALLISON MODEL 301 POWER SECTION
14,000 RPM - 1955°R

WESTINGHOUSE MODEL 19XB COMPRESSOR
14,000 RPM

MAXIMUM TIP JET THRUST AT
ZERO FLIGHT VELOCITY

PILOT & PASSENGER WEIGHTS
No. 10000. Weight per person 160 lbs.
Total weight 3200 lbs.



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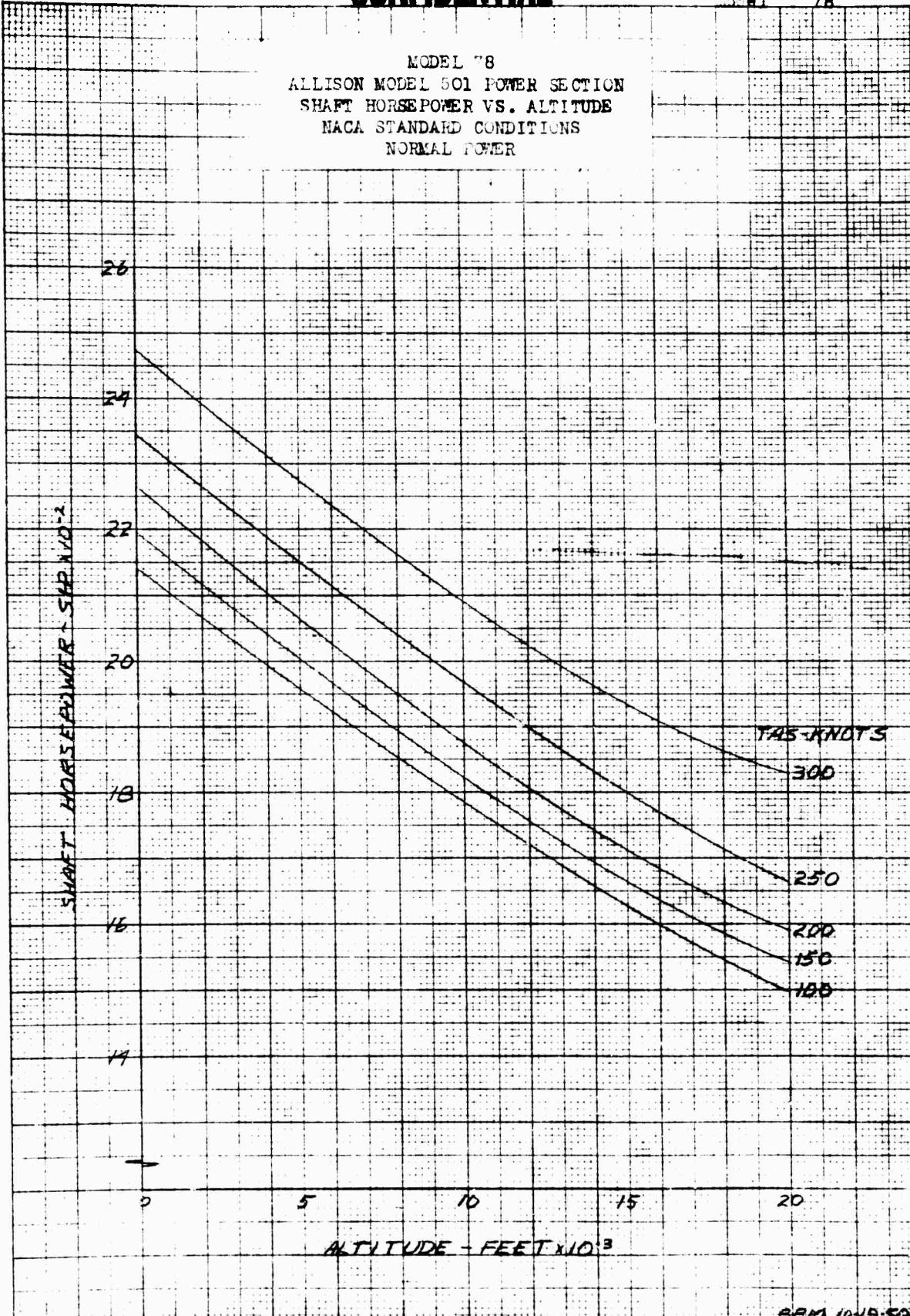
FIG 13

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MODEL 78
ALLISON MODEL 501 POWER SECTION
SHAFT HORSEPOWER VS. ALTITUDE
NACA STANDARD CONDITIONS
NORMAL POWER



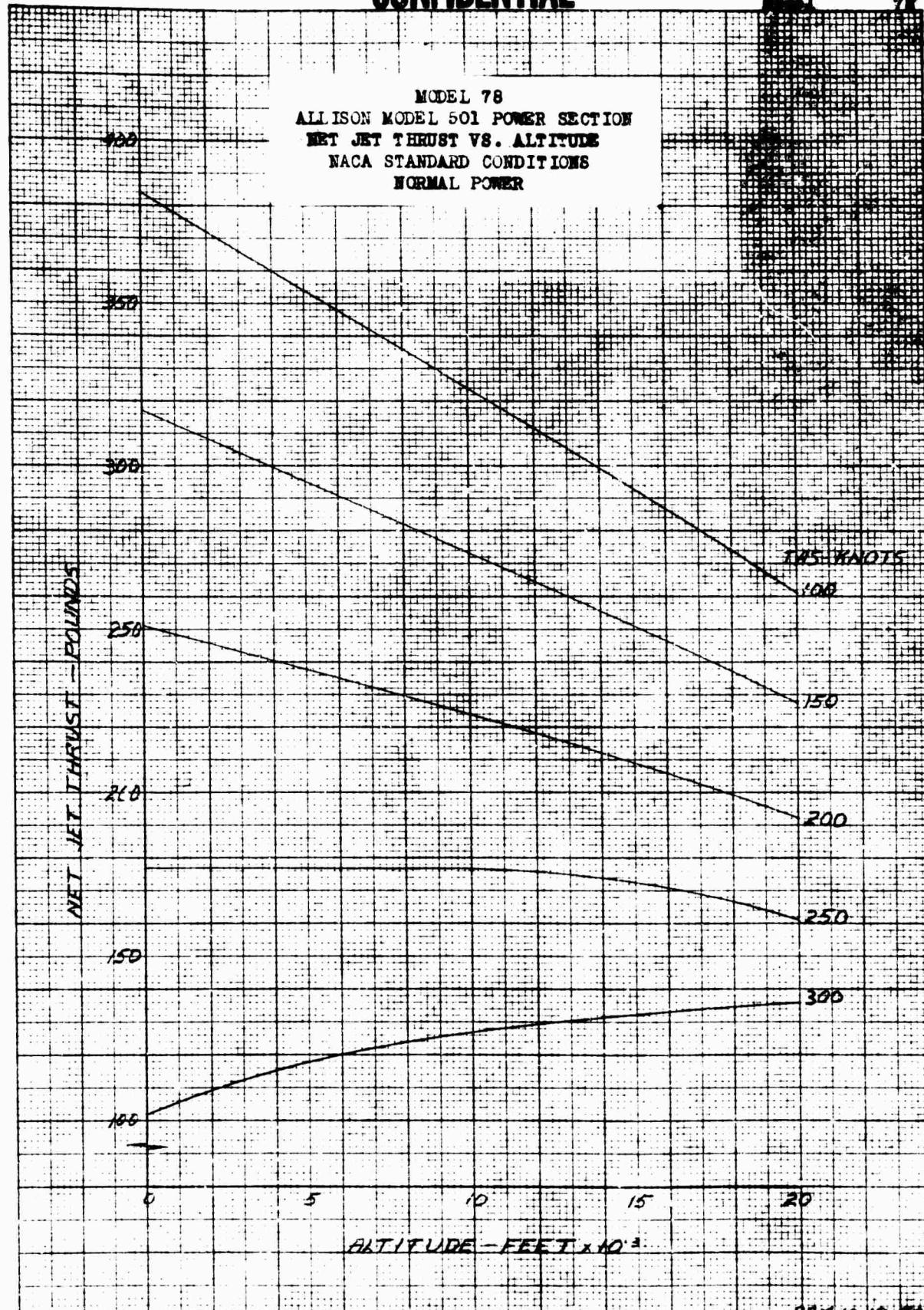
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MODEL 78
ALLISON MODEL 501 POWER SECTION
NET JET THRUST VS. ALTITUDE
NACA STANDARD CONDITIONS
NORMAL POWER



KODAK SAFETY FILM CO., N.Y. NO. 300-11
10 X 10 in. 16 lbs. 5th time mounted
KODAK SAFETY FILM
KODAK SAFETY FILM

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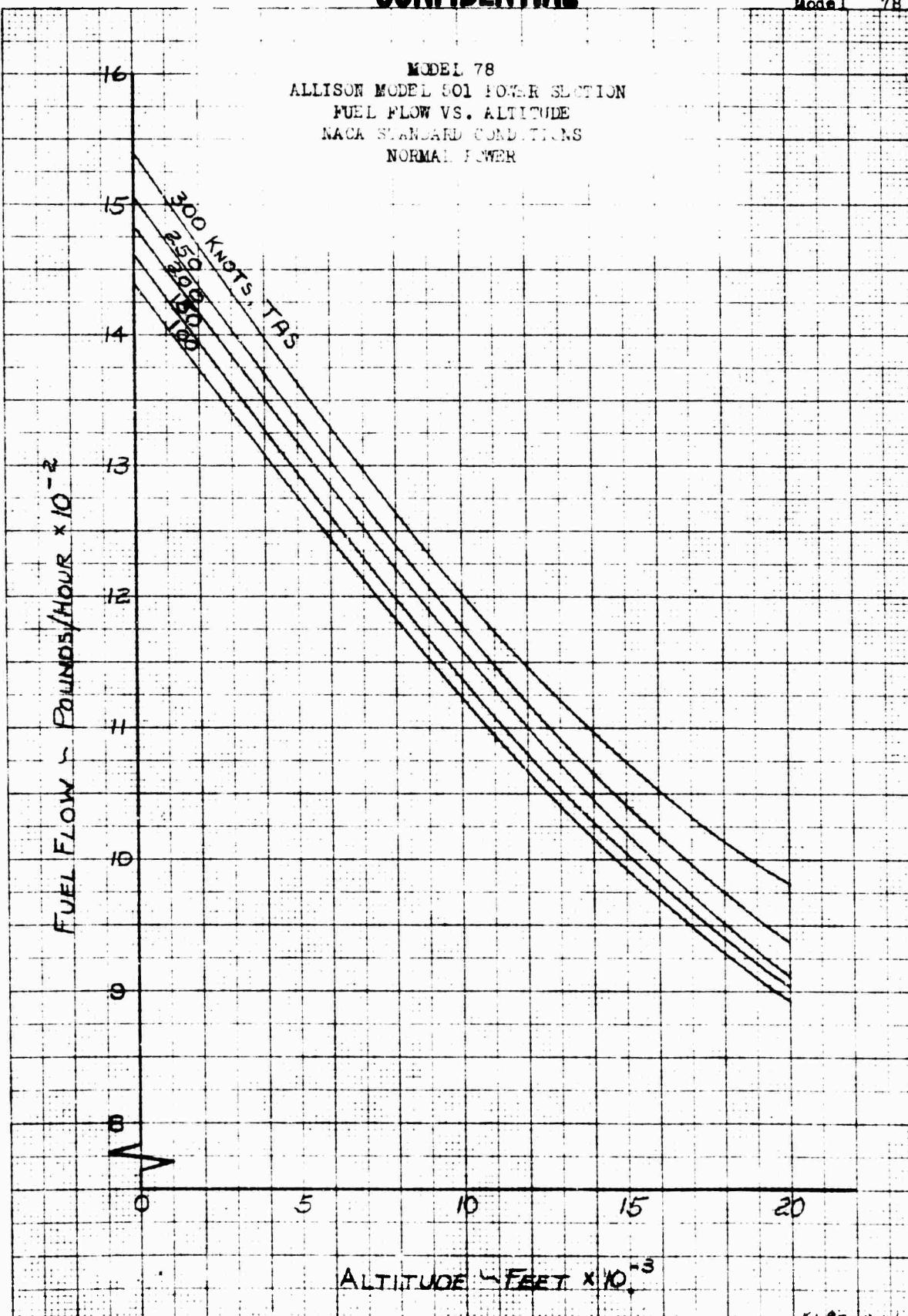
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FIG. 15

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FIG. 16

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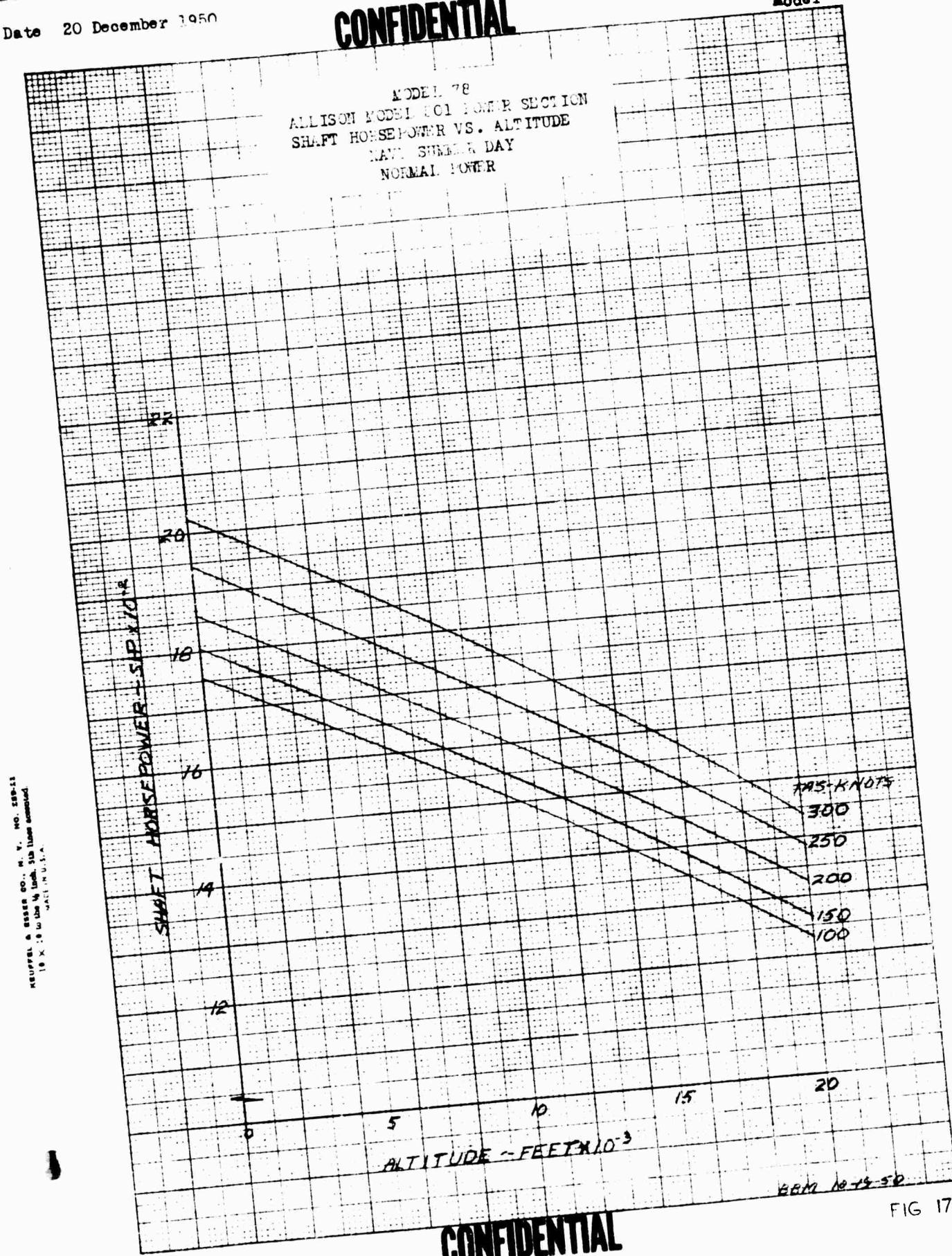


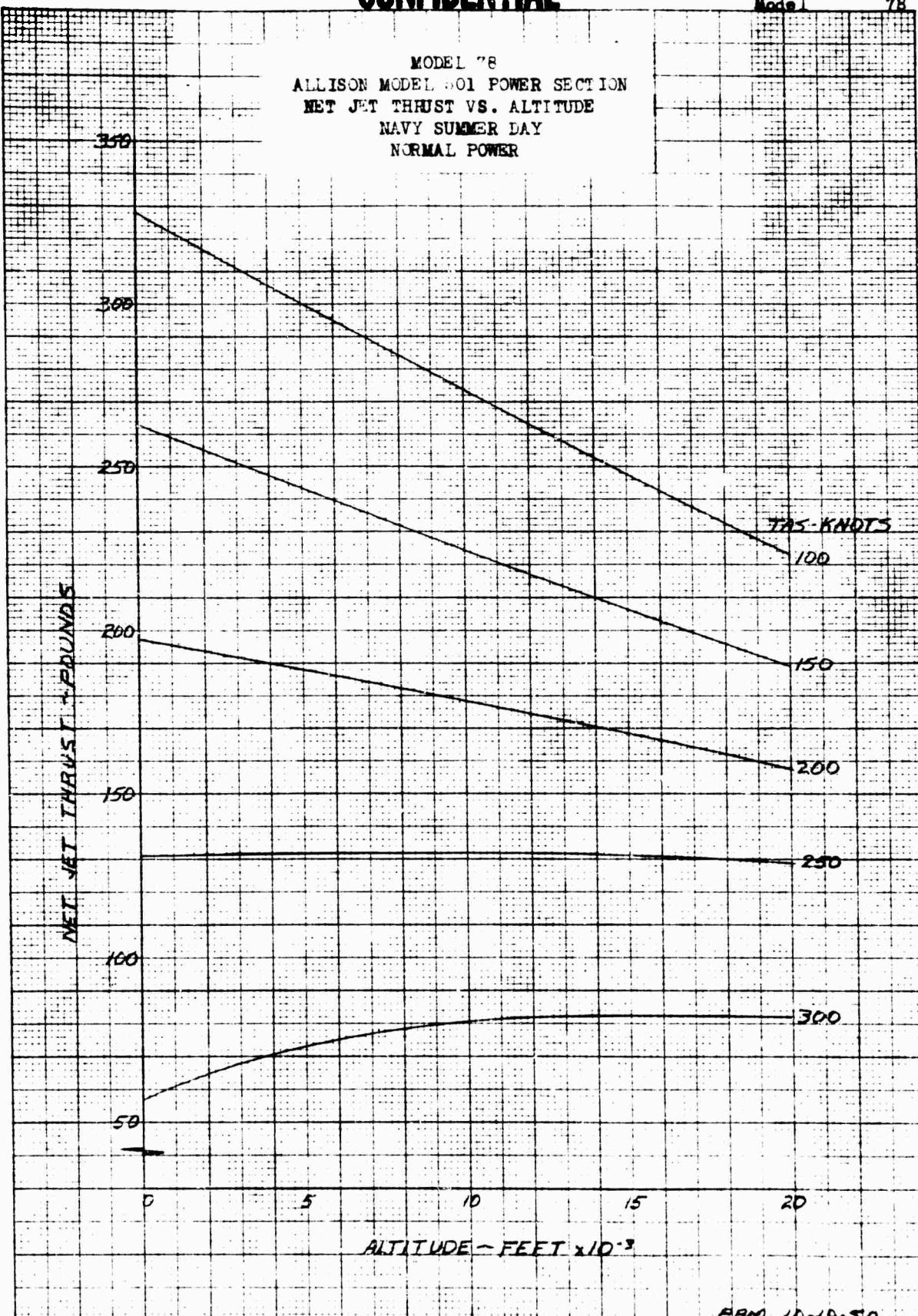
FIG. 17.

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MODEL "8"
ALLISON MODEL 301 POWER SECTION
NET JET THRUST VS. ALTITUDE
NAVY SUMMER DAY
NORMAL POWER



MAURER & SONS CO., N.Y. NO. 388-11
10 x 10 to the 1/4 scale (1st line horizontal)
MAURER U.S.A.

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FIG. 18

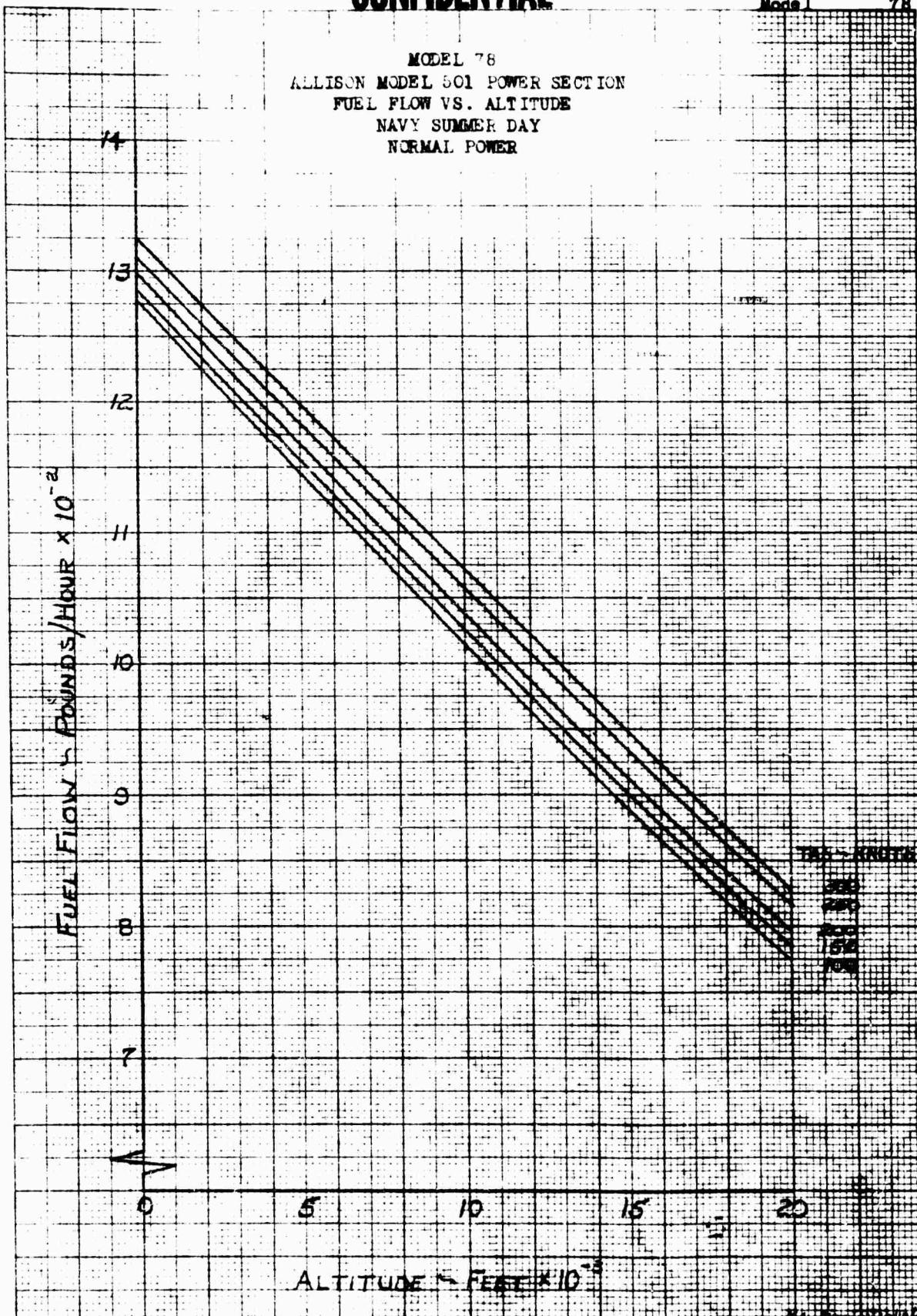
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MODEL 78
ALLISON MODEL 501 POWER SECTION
FUEL FLOW VS. ALTITUDE
NAVY SUMMER DAY
NORMAL POWER

KOUPPE & BESSER CO., NEW YORK
10 X 10 To The $\frac{1}{2}$ Inch Scale
MADE IN U.S.A.



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FIG. 19

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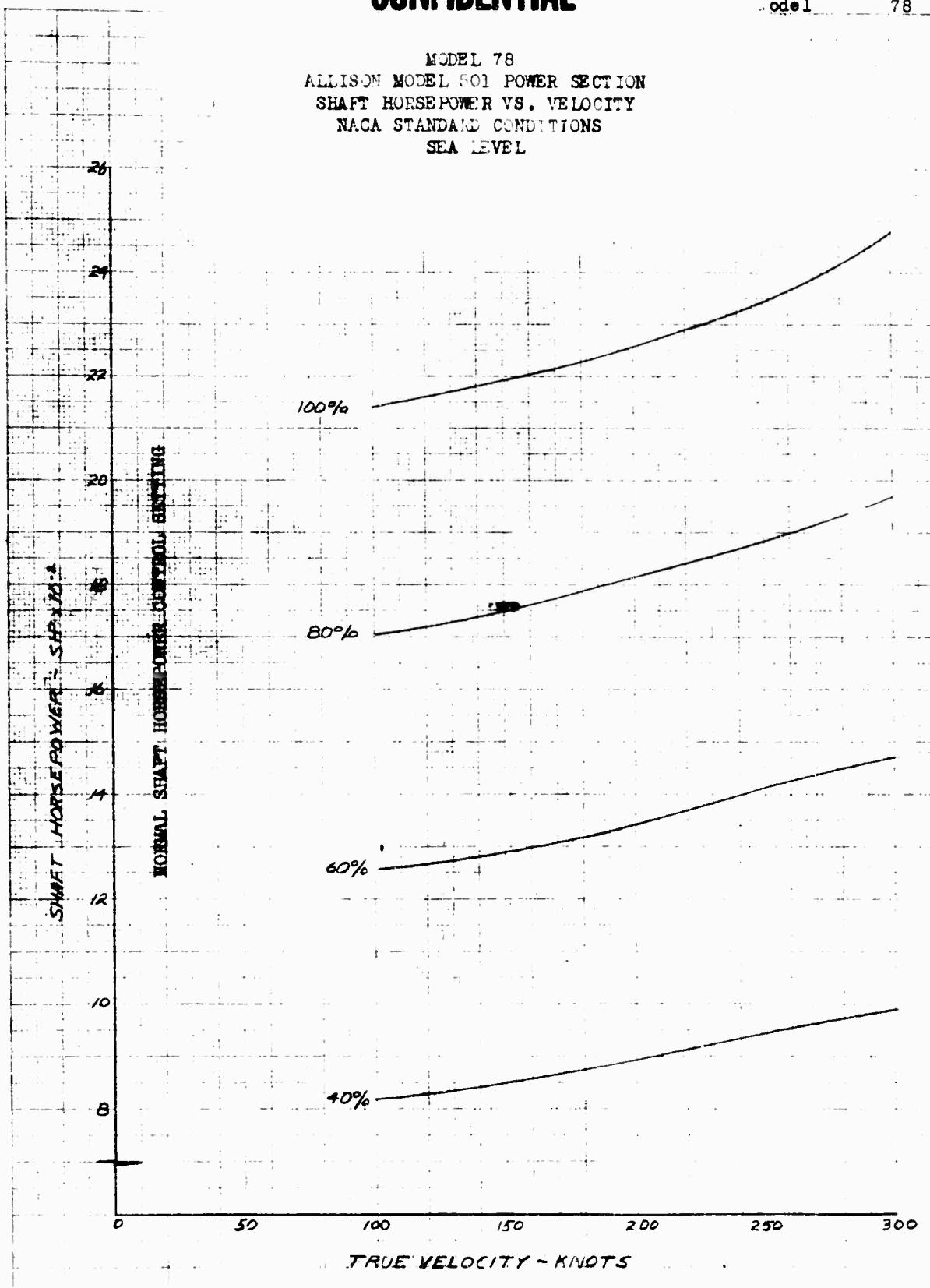
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MODEL 78
ALLISON MODEL 501 POWER SECTION
SHAFT HORSEPOWER VS. VELOCITY
NACA STANDARD CONDITIONS
SEA LEVEL

FEARREL & EISLER CO.

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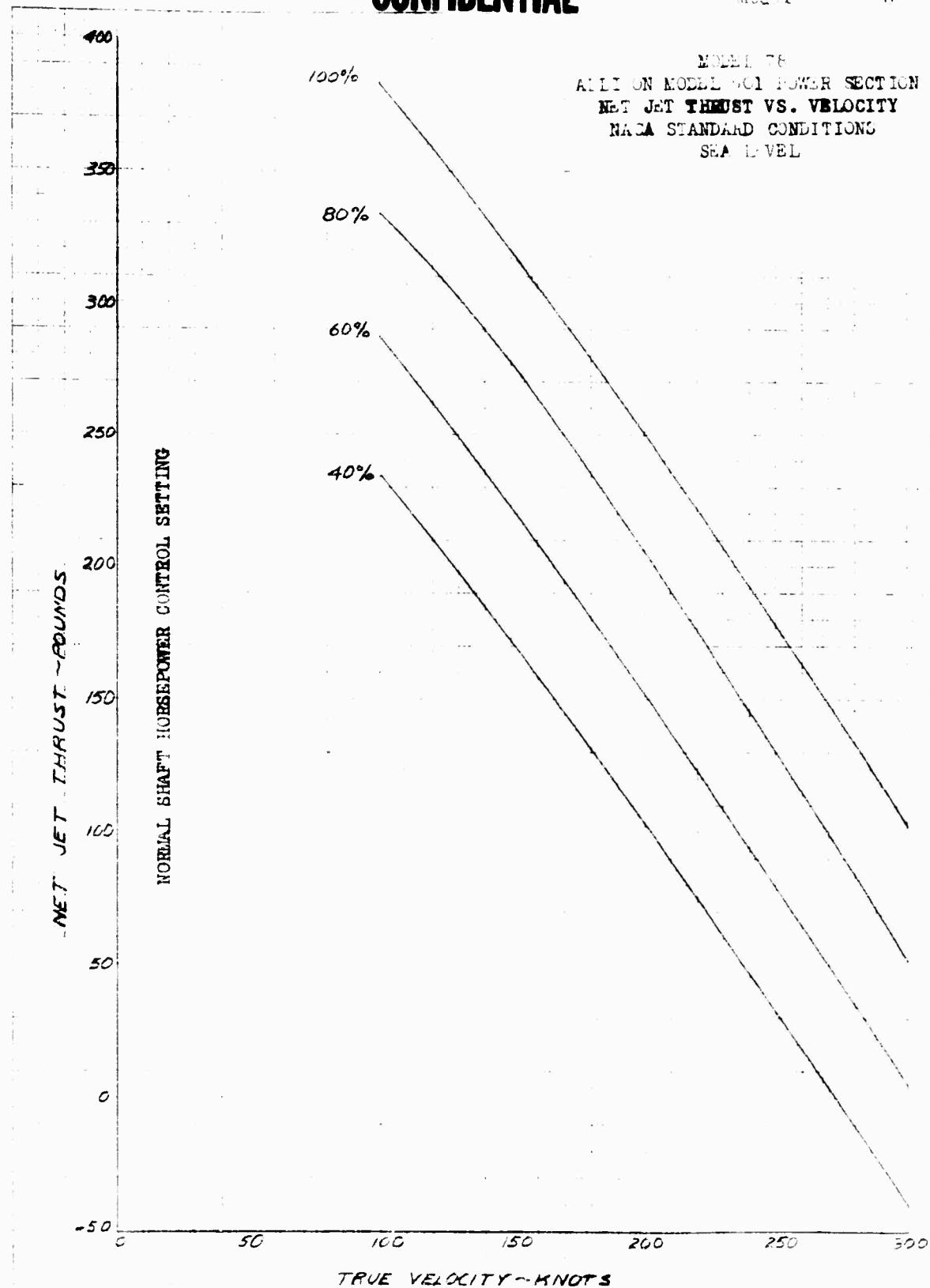
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FIG. 20

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ALLI ON MODEL 601 POWER SECTION
NET JET THRUST VS. VELOCITY
NACA STANDARD CONDITIONS
SEA LEVEL



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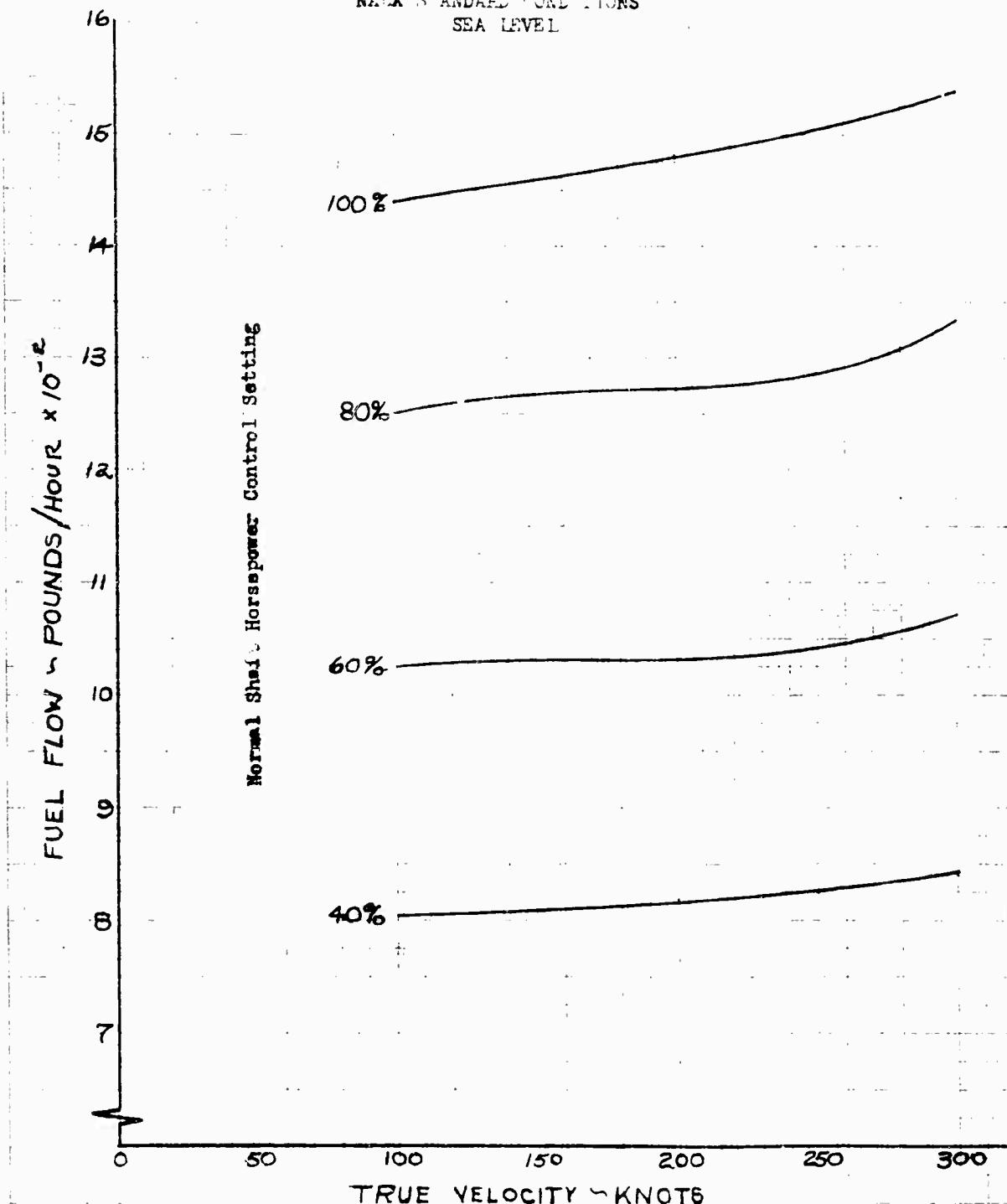
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FIG. 21

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ALLISON MODEL 301 POWER SECTION
FUEL FLOW VS. VELOCITY
NACA STANDARD CONDITIONS
SEA LEVEL



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Kil Brown 10/19/60

FIG. 22

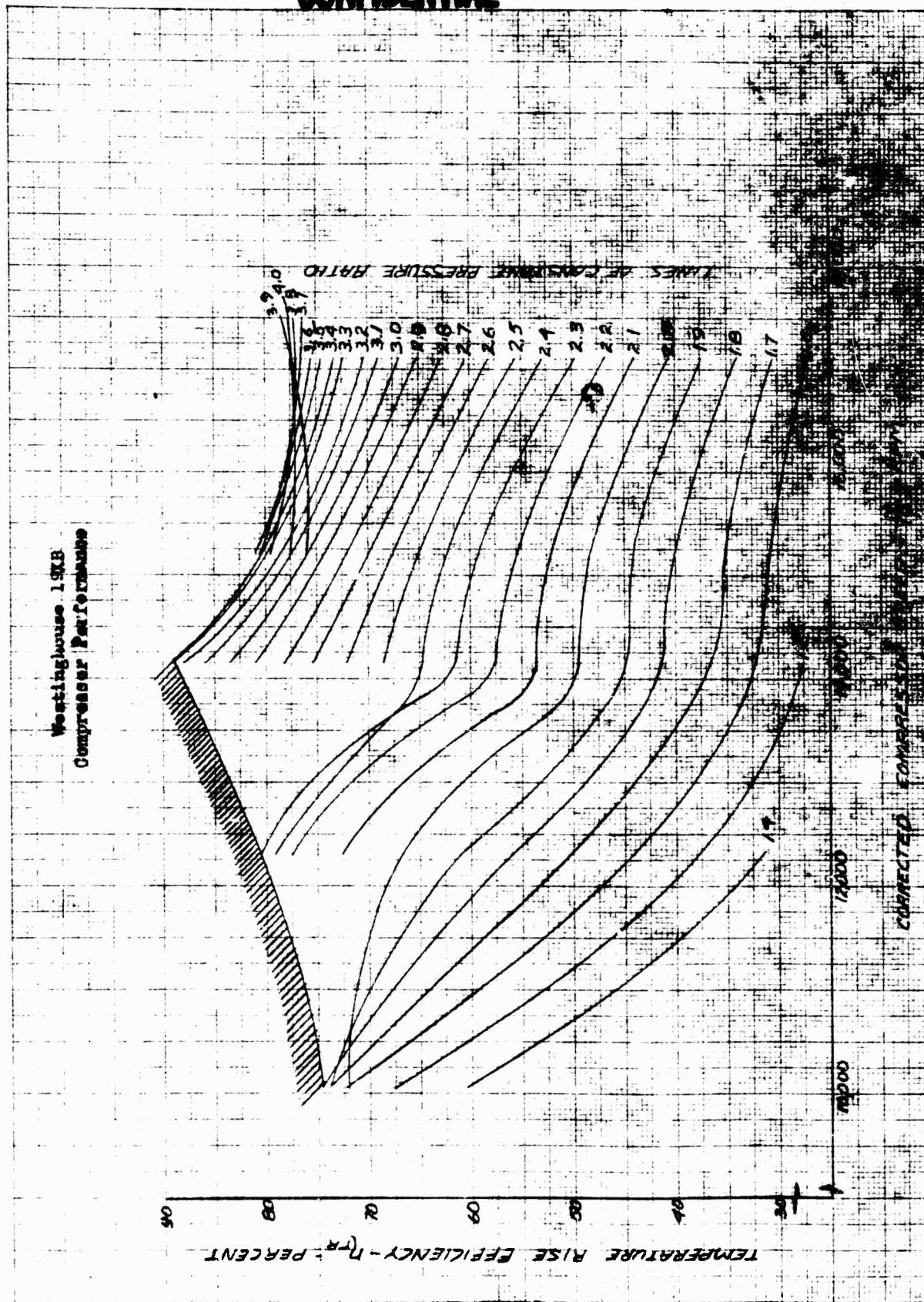
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FIG. 23

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WESTINGHOUSE 1905 COMPRESSOR
PRESSURE RATIO VS CORRECTED AIR FLOW

450

350

250

COMPRESSION PRESSURE RATIO
 R_p/R_1

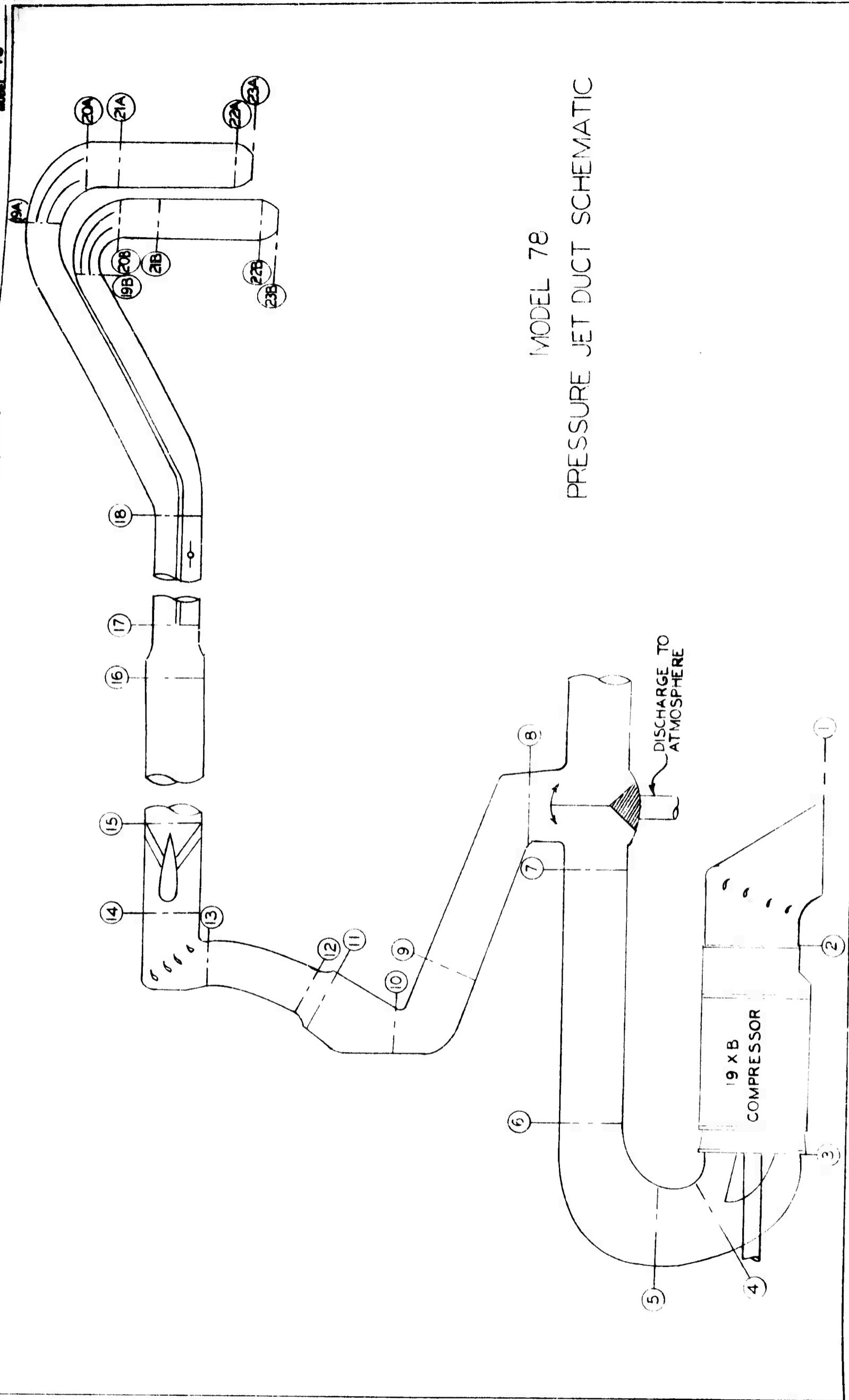
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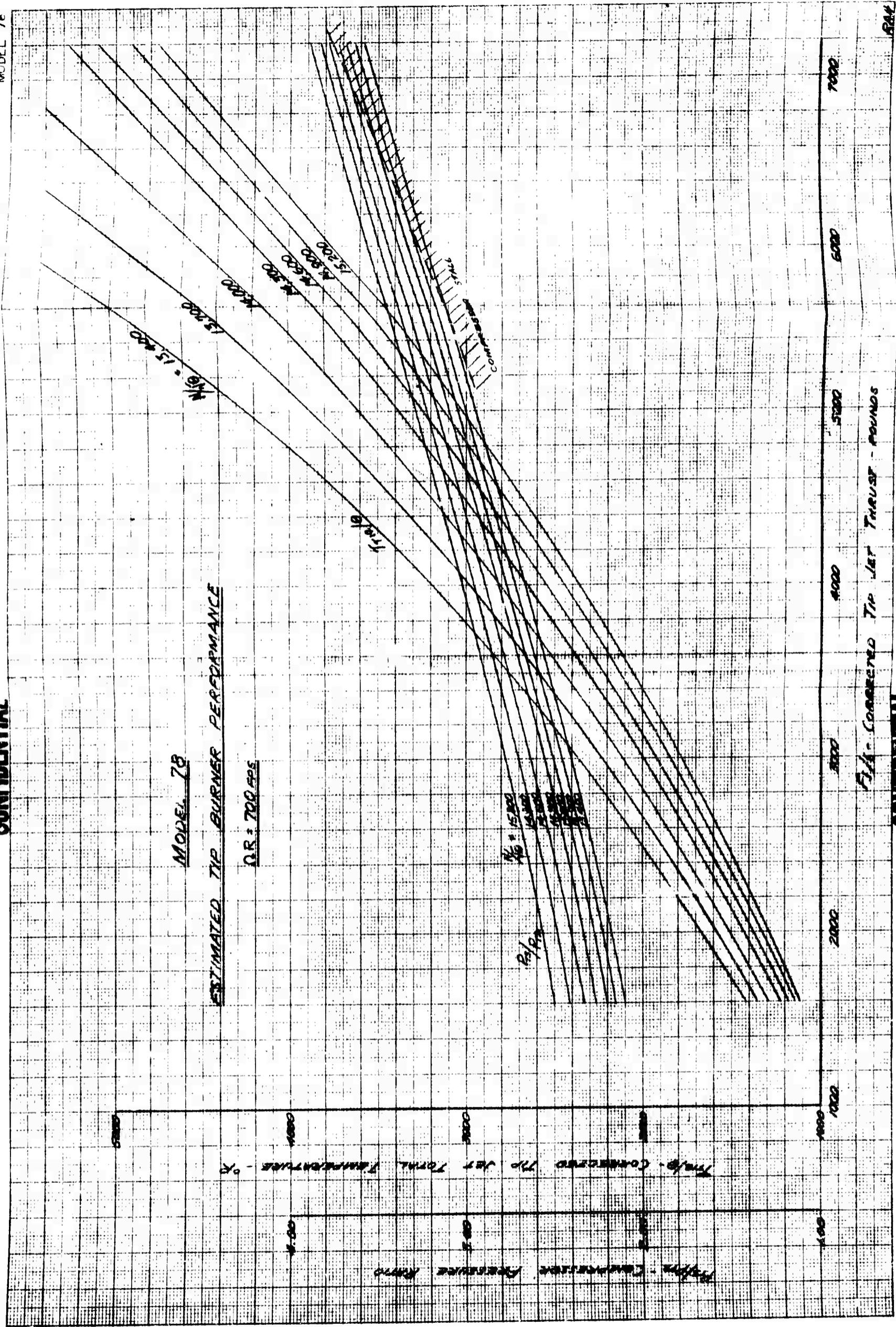
MODEL 78
PRESSURE JET DUCT SCHEMATIC

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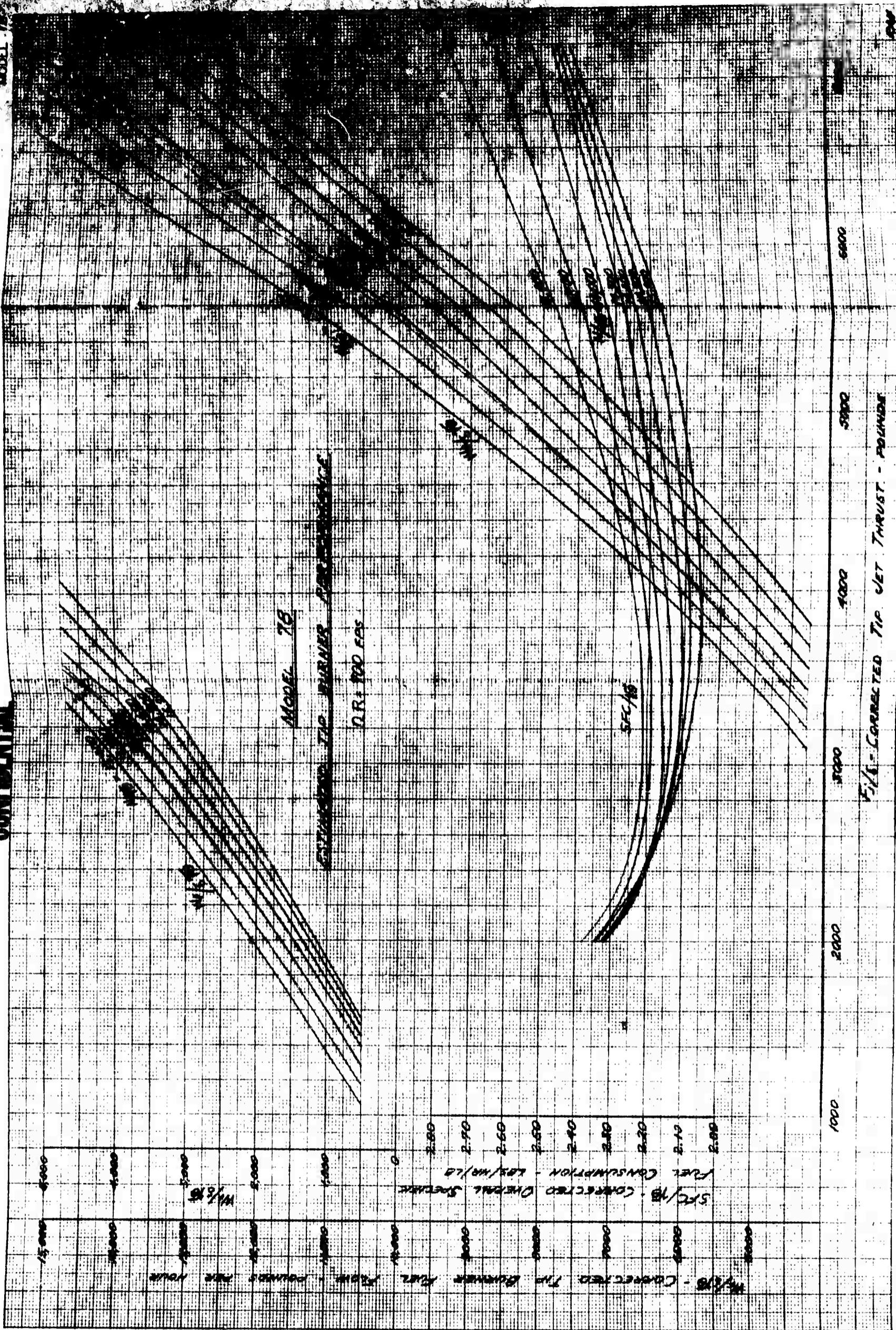
MACC 3150 (REF ID: K6-49)
FIG. 25

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MODEL 78

6000

TIP BURNER PERFORMANCE

NACA STANDARD DAY

100% RATED Power
14,000 RPM

T_{HE} = 4000°R
QR = 700 FPS

KELVIN & CELSIUS

No. 25011 10-1000 2-7 100

10000
8000
6000
4000
2000

ALTITUDE - FEET

TAS - KNOTS

150
100
50
0

0 5000 10000 15000 20000

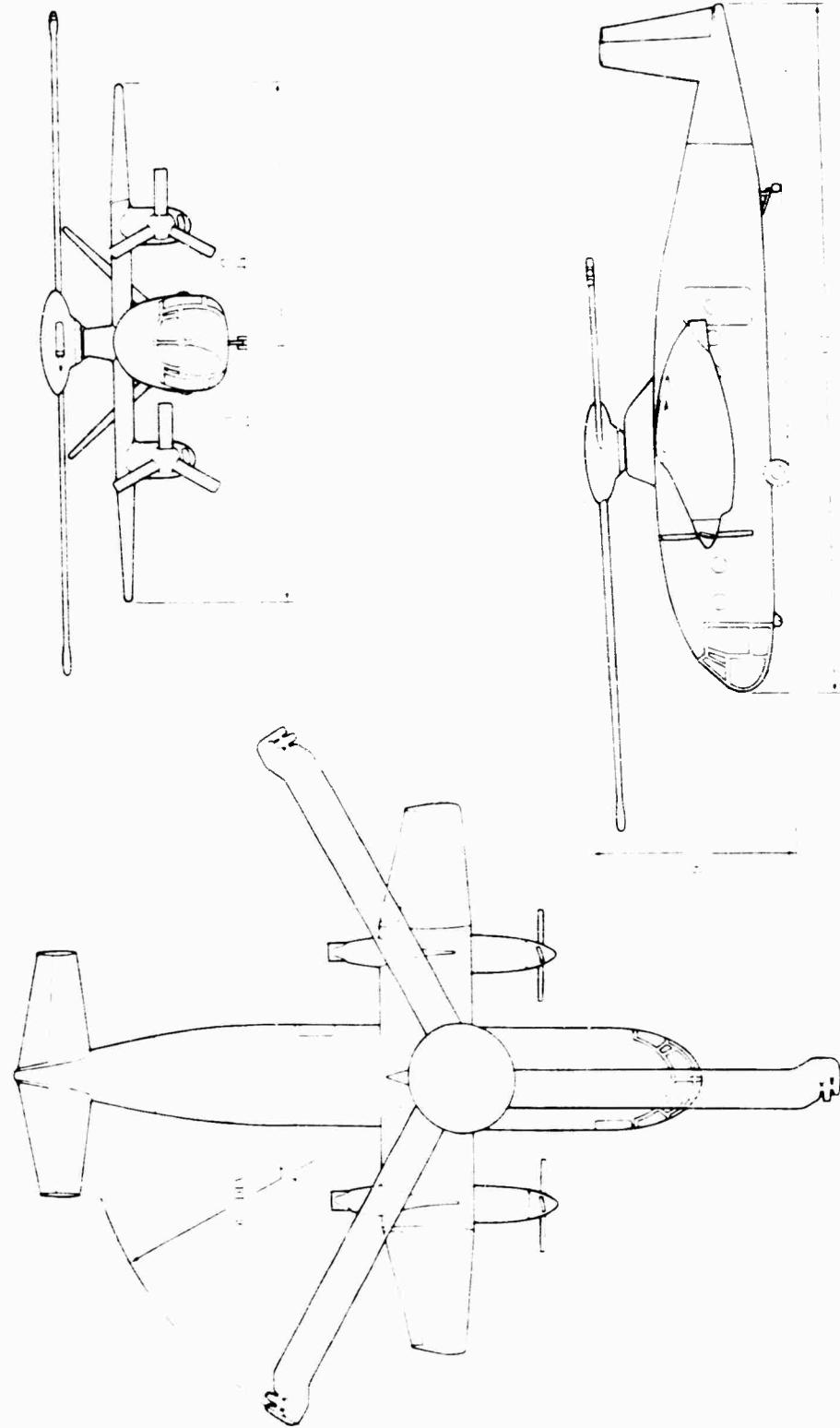
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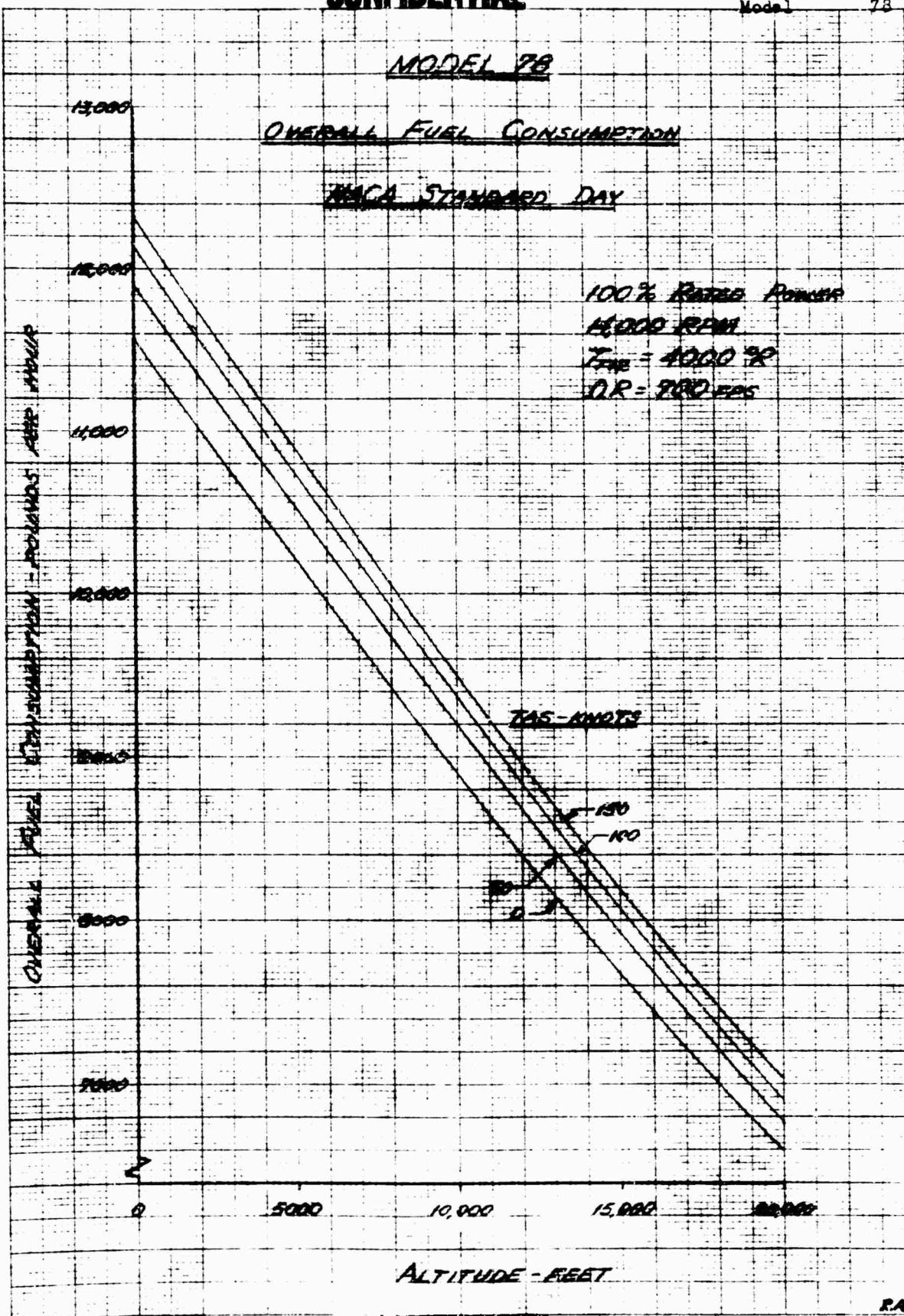
MODEL 78 GENERAL ARRANGEMENT

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FIG. 29

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MODEL 78

TIP BURNER PERFORMANCE

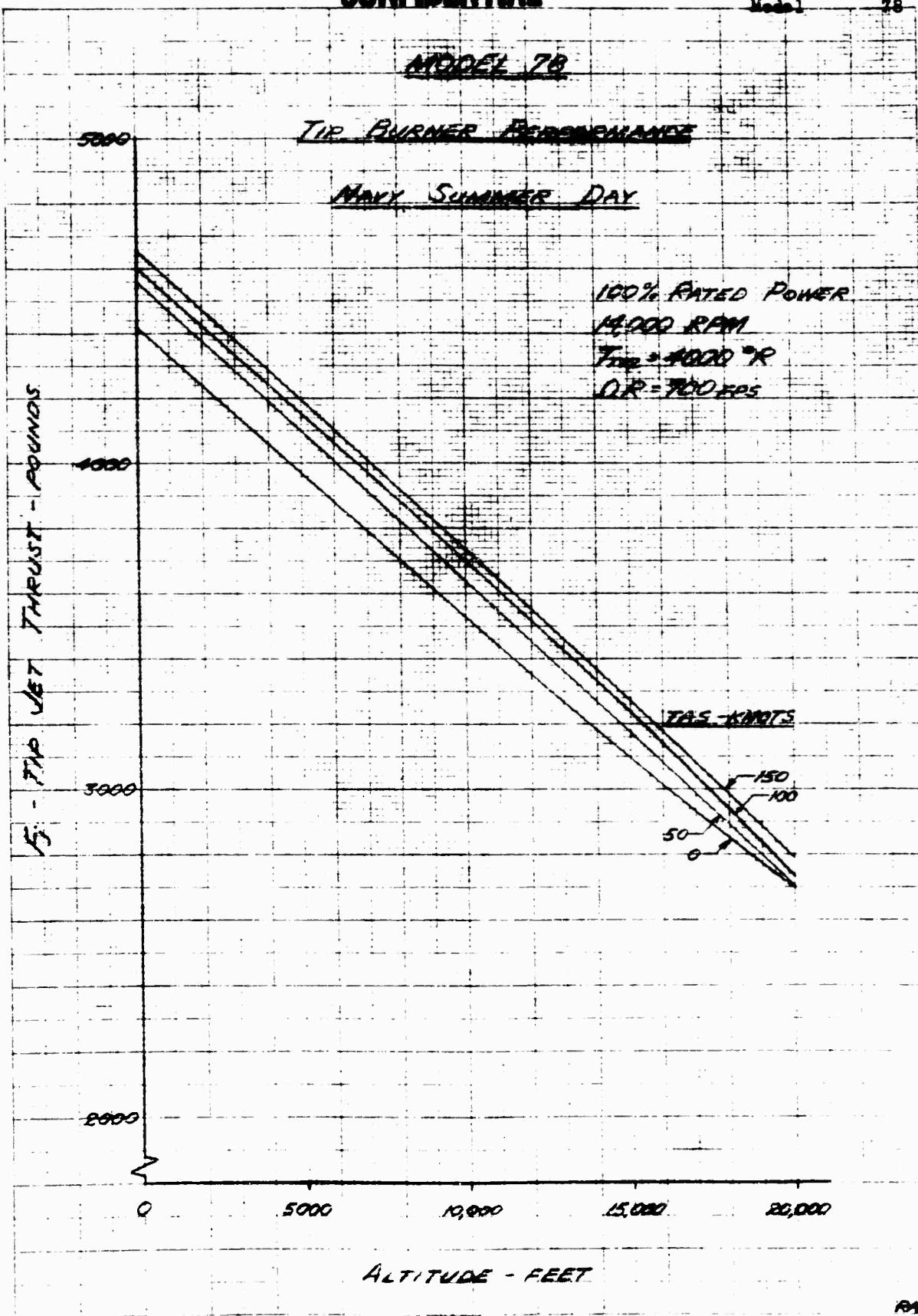
MARY SCHUMMER DAY

100% RATED POWER

14,000 RPM

T_{0,0} = 1020°F

DR = 720 LBS



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PAW
FIG. 20

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MOTSEY 58

OVERALL FUEL CONSUMPTION

MARY SUMMER DAY

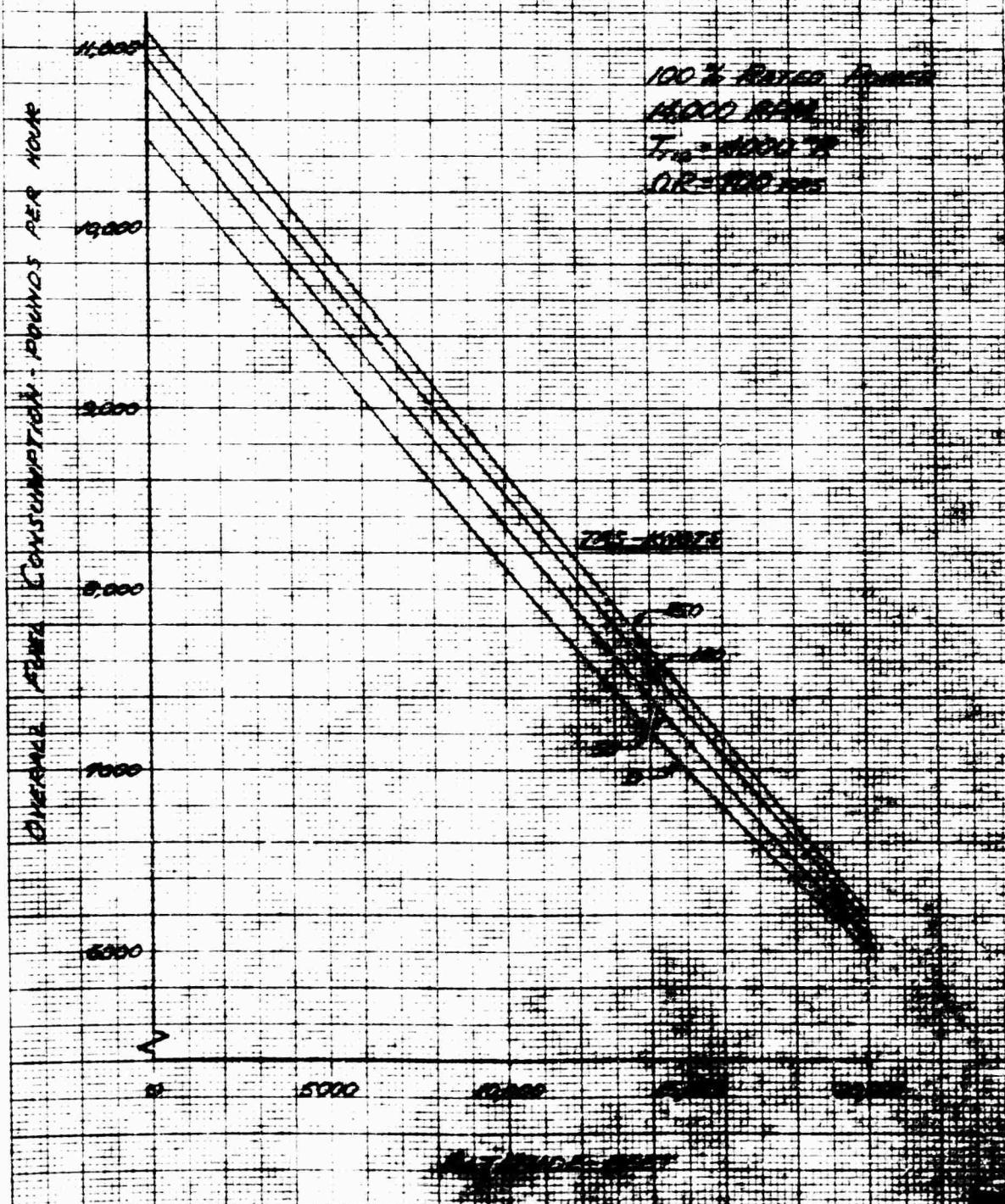


FIG. 31

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FIG. 2

ALLISON MODEL AND INLET SECTION INLET DUCT ANALYSIS

Quantity	V, ft/sec.	87° Knots	Seas.	NACA Standard Data	Normal Power
Total temperature, \bar{T}	0	1	1.1	1.30	1.45
Static pressure, \bar{P}_s , in. Hg.	20.2	20.2	25.2	20.2	20.2
Density, ρ , lb./cu. ft.	0.002372	0.002372	0.00219	0.00219	0.00215
Mass flow rate, \dot{m} , lb./sec.	0.14	0.14	0.19	0.19	0.19
Volume flow rate, \dot{V} , cu. ft./sec.	-	-	426	420	433
Cross-sectional area, A , sq. ft.	-	-	1.11	1.11	1.11
Velocity, V , ft/sec.	26.07	369	378	383	390
$\frac{1}{2} \rho V^2$, lb./sq. ft.	1.7	1.7	1.7	1.7	1.7
Velocity of sound, c , ft/sec.	1120	1110	1108	1112	1108
Mach number	0.151	0.334	0.340	0.292	0.312
Compressibility factor	1.000	1.000	1.03	1.022	1.032
Impact pressure, \bar{P}_i , in. Hg.	1.6	1.1	1.5	1.9	1.6
Total pressure, \bar{P}_t , in. Hg.	21.404	21.164	20.52	20.46	20.36
Change in total pressure, ΔP_t , in. Hg.	0	-0.4	-1.6	-1	-6
Change in static pressure, ΔP_s , in. Hg.	-0.6	-0.6	-0.6	-0.6	-0.6
Change in total temperature, ΔT	0	0	0	0	0
Pressure-loss coefficient, $\Delta \bar{L}/4c$	-	-	0.1	0.06	0.18
Velocity, V , ft/sec.	30.6	30.6	30.6	30.6	30.6
Static temperature, T_s , °F.	70.2	70.2	70.2	70.2	70.2
Air flow rate, \dot{m} , lb./sec.	0.14	0.14	0.14	0.14	0.14
Velocity, V , ft/sec.	3.0	2.72	2.68	3.20	3.20
Temperature ratio	1.02	0.95	1.024	1.016	1.017
Pressure ratio	1.02	0.95	1.027	1.017	1.019

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DATA: 54
REF ID: 190
MODEL: 78

TABLE 2
SECTION INLET DUCT ANALYSIS

Condition: T = 24° Rots., S = 1, NACA Standard Da., Normal cover

Quantity	Station	Station	Station	Station	Station	Station	Station
Total temperature, T_R	0	1	1.0	1.30	1.45	1.50	2
Static pressure, lb./sq. in.	21.0	21.0	21.0	21.0	21.0	21.0	21.0
Density, slugs/cu. ft.	0.02378	0.02411	0.02422	0.0229	0.0233	0.0231	0.0211
Mass flow of air, slugs/sec.	0.02	0.02	0.02	0.02	0.02	0.02	0.02
Volume flow of air, cu. ft./sec.	40.	44.6	42.6	42.0	42.4	46.4	
Cross sectional area, sq. ft.	A	1.01	1.01	1.01	1.01	1.01	1.01
Velocity, ft./sec.	V	40.	36.0	38.5	32.6	32.9	41.8
$\frac{1}{2} \rho V^2$ lb./sq. ft.	Q	194	162	185	124	125	185
Velocity of sound, ft./sec.	a	1120	1120	1120	1120	1120	1120
Mach number	M	0.02	0.02	0.02	0.02	0.02	0.02
Compressibility factor	F _c	1.02	1.025	1.031	1.011	1.022	1.035
Impact pressure, lb./sq. ft.	q _{F_c}	100	173	175	127	128	192
Total pressure, lb./sq. ft.	H	231	231	221.5	220.5	220.0	206.0
Change in total pressure, lb./sq. ft.	ΔH	0	-0.5	-17	-88	-64	-140
Change in static pressure, lb./sq. ft.	ΔP	33	-90	-19	39	-7	-64
Change in total temperature	ΔT_T	0	0	0	0	0	0
Pressureless coefficient	$\Delta H/c$	0.02	-0.1	0.05	0.05	1.05	
Velocity parameter	RW _c /m _A	0.654	0.573	0.563	0.590	0.592	0.736
Static temperature, R	T	518.4	520.0	522.7	525.3	522.1	522.6
Air flow rate, lb./sec.	W _c	31.0	31.0	31.0	31.0	31.0	31.0
Velocity, ft./sec.	V	370	370	385	325	328	415
Temperature ratio	T _r /T	1.02	1.024	1.027	1.023	1.016	1.017
Pressure ratio	P _r /P ₁	1.095	1.070	1.07	1.065	1.057	1.094

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TABLE 4
SUMMARY OF AIRLISCH "ODE" FOR POWER SECTION INLET DUCT ANALYSIS
See Level, Normal Power

Condition	Static	$V_C = 87$ Knots	$V_C = 240$ Knots
Velocity ratio V_D/V_C	~	2.01	0.803
Angle of Attack α	0	0	0
Estimated E/q_c	1.11	0.815	1.04
Estimate of losses through the duct			
Condition	Static	$V_C = 87$ Knots	$V_C = 240$ Knots
Section 0-1	1-1.1	1.01-1.3	1.04-1.5
$\Delta H/q_{local}$	0.800	0.0016	0.0522
q_{local}/q_c	0.9150	0.9570	0.7010
$\Delta H/q_c$	0.4390	0.0876	0.0494
Total q_{tot}/q_c		1.0126	0.8131

Overall pressure loss and effect upon engine performance

Condition	Static	$V_C = 87$ Knots	$V_C = 240$ Knots
$\Delta H_{tot}/q_c$	1.0126	0.8131	0.8111
$\Delta H_{tot}/R_{tot}$		0.0784	0.0611
$\Delta SEP/SEP$	0.126		0.0622
$\Delta F/F$		0.122	0.0791
$\Delta W_F/W_F$	0.0644		0.0357

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TABLE 5WEETING U.S.H. 1951 CORRECTION

Condition: Static, Sea Level, NASA Standard Day, 15°, 1013.25 mb

Quantity	Symbol	0	1	1.1	2
1 Total temperature, °F	T _T	510.4	511.4	510.4	510.4
2 Static pressure, lbs./sq.ft. abs.	P	14.70	14.70	14.70	1782
3 Density, slug/cu.ft.	ρ	.001675	.001675	.001675	.002077
4 Mass flow of air, slug/sec.	m	.754	.754	.754	.754
5 Volume flow of air, cu.ft./sec.	q		300	300	300
6 Cross-sectional area, sq.ft.	A	.250	.250	.250	.250
7 Velocity, ft. sec.	V		300	300	300
8 $\frac{1}{2}V^2$, lbs./sq.ft.	q		100	143	100.0
9 Velocity of sound, ft. sec.	c	1110	1110	1110	1110
10 Mach number	M		.317	.317	.317
11 Compressibility factor	F _C		1.027	1.027	1.027
12 Impact pressure, lbs./sq.in.	q _P		14.7	14.7	14.7
13 Total pressure, lbs./sq.in.	P _T	1110	1110	2043	1970
14 Change in total pressure, lbs./sq.in. ΔP			-43	-44	-44
15 Change in static pressure, lbs./sq.in. ΔS			-1	-1.0	-1.0
16 Change in total temperature, °F ΔT_T			-	-	-
17 Pressure-loss coefficient $\Delta h'/q$			-	-	-
18 Velocity parameter R_{∞}/V_{∞}		.647	.44	.44	.38
19 Static temperature, °F T_s	T _s	51.4	51.4	51.4	51.4
20 Airflow rate, cu. sec.	a	24.	24.0	24.0	24.0
21 Velocity, ft. sec.	V		300	300	300
22 Temperature ratio T/T_s		1.117	1.021	1.021	1.021
23 Pressure ratio P/P_s		1.0	1.0	1.0	1.0

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TABLE 7 - ANALYSIS OF PRESSURE JET SYSTEM

$$N\sqrt{V_e} = 14,000 \quad \frac{T_3}{T_2} / \frac{P_3}{P_2} = 2.40$$

Station Qty.	Unit	0	3	4	5	6	7	8
W_a	#/sec	28.45	28.45	28.45	28.45	28.45	28.45	56,80
T_T	°R	518.4	740	740	740	740	740	740
P_T	PSIA	14.7	35.28	35.18	34.96	34.77	34.59	34.37
A	in ²		187	100	100	100	100	200
RW_a/P_{TA}		.230	.431	.434	.436	.438	.441	
V	ft/sec	170	328	331	333	334	337	
T	°R	737.62	731.13	730.97	730.86	730.81	730.75	
γ		1.3942	1.3944	1.3944	1.3944	1.3944	1.3944	1.3945
$\gamma/\gamma - 1$		3.5368	3.5355	3.5355	3.5355	3.5355	3.5349	
T_T/T		1.00323	1.01213	1.01235	1.01251	1.01258	1.01266	
P_T/P		1.01147	1.04360	1.04440	1.04487	1.04512	1.04540	
P	PSIA	34.88	33.71	33.47	33.28	33.10	32.88	
q_e	PSI	0.40	1.47	1.49	1.49	1.49	1.49	1.49
f	ft	2.95*	2.95	2.95	2.95	2.95	2.767	
RN		2.45×10^6	3.1×10^6					
f				.0031				
$1/D$				10.15				
$\Delta H/q$.25	.15	.120	.12	.10	.05
ΔH			.10	.22	.19	.13	.22	.07
V_{Tx}								
V_{Ty}								
• $V_{Ty} - V_{Tx}$								
P_{Ty}/P_{Tx}								
ΔH								

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TABLE 7 Continued

Qty.	Station Unit	9	10	11	12	13	14	15
W _a	#/sec	56.90	56.90	56.90	56.90	18.97	18.97	18.97
T _T	°R	740	740	740	740	740	740	740
P _T	PSIA	34.30	34.18	33.87	33.83	33.80	33.53	33.23
A	in ²	200	209	410	251	78.4	44.2	44.2
RW _a /P _T A		.442	.425	.218	.357	.382	.682	.688
V	ft/sec	338	323	162	268	288	551	557
T	°R	730.58	731.40	737.84	734.08	733.17	715.96	714.41
γ		1.3945	1.3944	1.3941	1.3943	1.3944	1.3950	1.3950
γ/γ -1		3.5349	3.5355	3.5374	3.5361	3.5355	3.5316	3.5316
T _T /T		1.01289	1.01176	1.00293	1.00806	1.00932	1.03358	1.03582
P _T /P		1.04632	1.04215	1.01040	1.02876	1.03334	1.1238	1.1323
P	PSIA	32.78	32.77	33.52	32.88	32.71	29.84	29.35
qc	PSI	1.02	1.38	0.35	0.95	1.09	3.69	3.88
ρ	ft	4.767			14.45			1.96
Wa/gp		.371			.122			.300
RN		3.1x10 ⁶			1.1x10 ⁶			1.45x10 ⁶
f					.0035			.0031
l/D					1.90			11.3
△ H/q		.10	.20	.10	.027	.25	.10	.14
△ H		.15	.28	.04	.05	.27	.37	.54
V _{T_x}						0		75
V _{T_y}						75		220
V _{T_y} - V _{T_x}						5625		42,775
P _{T_y} /P _{T_x}						1.00223		1.01698
△ H							.07	

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TABLE 7 Continued

Qty.	Station Units	16	17a	18a	19a	20a	21a	22a
W _a	#/sec	18.97	9.48	9.48	9.49	9.48	9.48	
T _T	°R	740	740	740	740	740	740	
P _T	PS IA	33.25	33.25	36.01	36.85	36.84	36.54	36.22
A	in ²	44.2	24.3	24.3	24.3	48.4		
RW _a /P _T A		.688	.625	.577	.564	.284		
V	ft/sec	557	498	453	441	212		
T	°R	714.41	719.55	723.08	723.97	736.30		
γ		1.3950	1.3948	1.3947	1.3947	1.3943		
γ/γ -1		3.5316	3.5329	3.5336	3.5336	3.5361		
T _T /T		1.03582	1.02842	1.02340	1.02214	1.00503		
P _T /P		1.1323	1.1040	1.0850	1.08035	1.01790		
P	PS IA	29.37	30.12	33.19	34.11	36.19		
q _c	PSI	3.88	3.13	2.82	2.74	.65		
P	ft		17.08					
W _a /q _p			.172					
RH				1.45x10 ⁶				
f				.0033				
1/D			44.0					
Δ H/g		.001		.58	.15	.15	2.0	
Δ z		0		1.82	.42	.41	1.30	.32
V _{Tx}			220	618	680			
V _{Ty}			618	680	700			
V _{Ty} - V _{Tx}			333,524	80,476	27,600			
P _{Ty} /P _{Tx}			1.1378	1.0250	1.01089			
Δ H		.56	0	4.58	1.26	.40		

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TABLE 7 Continued

Station	23a	17b	18b	19b
Qty. Units				
W _a #/sec		9.48	9.48	9.48
T _r °R	1885	740	740	740
P _T PSIA	35.22	33.25	35.89	36.62
A in ²	22.31	22.1	22.1	22.1
RW _a /P _T A		.688	.637	.624
V ft/sec	1932	557	507	.497
T °R	1594	714.41	718.80	719.63
γ		1.3950	1.3949	1.3948
γ/γ -1		3.5316	3.5323	3.5329
T _T /T		1.03582	1.02949	1.02631
P _T /P		1.1823	1.1060	1.1034
P PSIA		29.37	32.39	33.19
q _c PSI		3.68	3.50	3.43
P ft		1.39		
W _a /gp		2.11		
RH		1.65x10 ⁶		
f		.0032		
1/D		39.34		
ΔE/q		.50	.15	.15
ΔE		1.94	.53	.51
V _{T_x}				
V _{T_y}				
V _{T_y} - V _{T_x}				
P _{T_y} /P _{T_x}		1.1378	1.0350	1.01089
Δ E		.4008	1.28	.40

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<u>Station</u>	<u>20b</u>	<u>21b</u>	<u>22b</u>	<u>23b</u>
Qty. Unit				
W_a #/sec	9.48	9.48		
T_T °R	740			1865
P_T PSIA	36.51	35.21	34.91	34.91
A in ²	48.4			22.51
$RW_a/P_T A$.280			
V ft/sec	213			1922
T °R	730.20			1576
γ	1.3943			
$\gamma/\gamma - 1$	3.6361			
T_T/T	1.00508			
P_T/P	1.01808			
P PSIA	35.86			
q_0 PSI	.65			
P ft				
W_a/q_0				
RN				
1/D				
$\Delta E/q$		2.0		
ΔE		1.30		.30
V_{T_x}				
V_{T_y}				
$V_{T_y} - V_{T_x}$				
P_{T_y}/P_{T_x}				
$\triangle E$				

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TABLE 7 Continued

Duct	a	b	Total Model 78
Qty. Unit			
A ft ²	.1563	.1563	469
P _{T22} PSIA	35.22	34.91	
γ	1.3649	1.3664	
γ/g -1	3.7405	3.7393	
(2/γ + 1) ^{γ-1}	.5348	.5345	
P ₂₃ PSIA	18.83	18.66	
W _E #/sec	9.60	9.1	
T ₂₃ °R	1594	1516	
T _T °R	1885	1865	1875
C _{pav}	.2676	.2671	
Δ T °R	858	840	
W _F #/sec	.12931	.12637	176704
W _F #/hr	465.5	454.9	2761
F/A	.01364	.01333	
V ft/sec	1932	1932	
η $\frac{W_F}{g} \Delta V$ #	350	346	
(Δ P)A #	93	89	
F _J #	443	443	2634
H _{req}			4290
H _{out put/engine}			2283
W _{FE} /engine			1480
W _{PE} total			2960
W _{FE} + W _F			3721
Overall SFC			2.17

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